University of Virginia

Two Designs for an Orbital Transfer Vehicle

USRA/NASA Advanced Design Program
Summer Conference June 1988

(NASA-CR-184697) TWO DESIGNS FOR AN ORBITAL TRANSFER VEHICLE. PROJECT ORION: A PROPOSAL FOR A MANNED ORBITAL TRANSFER VEHICLE FOR THE 21ST CENTURY (Virginia Univ.) 129 p N89-20176

CSCL 22B G3/16 Unclas 0189667

Report 2 of 3

PROJECT ORION

A Proposal
For a Manned Orbital Transfer Vehicle
For the 21st Century

Submitted by WWSR Incorporated

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Important Abbrevations

AOTV	Aerobraked (Aeroassisted) OTV
CCM	Crew Command Module
c.g.	center of gravity
ECLSS	Environmental Contol & Life Support Systems
EMU	Extra-vehicular Mobility Unit
EPS	Electrical Power Systems
EVA	Extra-vehicular Activity
EVAM	EVA Module
GEO	Geosynchronous Orbit
GNC	Guidance, Navigation and Control
LEO	Low Earth Orbit
LH2 or LH_2	Liquid Hydrogen
LO2 or LO_2	Liquid Oxygen
MLI	Multi-layered Insulation
MMU	Manned Maneuvering Unit
OMV	Orbital Maneuvering Vehicle
OTV	Orbital Transfer Vehicle
PIB	Phased Injection Burn
RCS	Reaction Control System
TPS	Thermal Protection System

Foreword

The following paper is the final report in fulfillment of the requirements for the undergraduate design sequence in aerospace engineering, AE 441-442. It is the culmination of nine months of work completed by Group W on its design for a manned orbital transfer vehicle.

Even though Group W did not work on a project for the AIAA design competition, we did attempt to present a proposal that would meet the requirements of the competition if a request for proposal had been made for our design. This meant that we needed not only to design an OTV but to address such concerns as costs, manufacturing, and management.

For this reason, the paper is written to be a proposal from an aerospace corporation that is to be presented to NASA. WWSR Inc. was created to be this fictious corporation. WWSR is a composite of many aerospace corporations. Any similiarities to an actual corporation is purely coincidential.

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Introduction

Since the late 1970s to early 1980s, there has been considerable research into the deployment of an American space station. The proposed Space Station will allow for a permament manned settlement in space. It will also permit numerous spaced-based missions that may not have been practical in the past. One of these missions is the deployment of an orbital transfer vehicle (OTV). The purpose of an OTV is to make excursions from one orbit to another. More specifically, it is to be capable of going into higher Earth orbits than the Space Shuttle. There is also a major difference between an OTV and its counterpart the orbital maneuvering vehicle (OMV) in that an OMV is only designed for orbit changes of a few hundred miles while the OTV is designed for orbit changes of thousands of miles. For the most part, current OTVs have been designed to be able to go, at the very least, from Low Earth Orbit (LEO) to a geostationary orbit (GEO). NASA has been investigating several proposals from other areospace firms for OTVs. A few proposed OTVs have been ground-based, but most have been designed to be permamently based at the Space Station.

In *Pioneering the Space Frontier: The Report of the National Commission on Space*, the Commission states that:

> A high priority exists for this vehicle [an OTV], which will greatly lower the cost of access to geostationary orbit and to the Moon for crews and payloads ranging from 10 to 20 tons. The transfer vehicle will be modular, single-stage, fueled by liquid oxygen and liquid hydrogen, and outfitted with an aerobrake to conserve fuel by allowing the vehicle to slow down through the drag of Earth's atmosphere... With appropriate modification the transfer vehicle could be used as a lunar lander [1, p. 122].

In response to the need for an OTV expressed in the report, WWSR has created a proposal for a manned OTV that meets the criteria selected by the Commission. The design that WWSR is proposing will also meet the following criteria:

1. Be based at the Space Station.
2. Have the capability of supporting 3 people for a mission lasting no longer than 14 days.
3. Be able to perform multiple missions between LEO and GEO with a minimum amount of servicing.
4. Carry a maximum payload of 24,000 pounds between LEO and GEO.
5. Support EVA.

The primary mission of the OTV is to support manned excursions to GEO to service a satellite in orbit without needing to return it to the Space Station or to Earth. WWSR realizes, however, that it may not be possible due to some unique failure of a satellite to repair it at GEO. For this reason, the OTV has been designed to be capable of bringing the satellite back to the Space Station. It is also capable of returning the same (or another) satellite to GEO. This eliminates unnecessary missions to GEO by other payload delivery systems (such as the PAM Centaur).

WWSR has based its design on a "worst case" scenario. This scenario is a mission that consists of the following:

1. Leaving the Space Station, going to GEO, and returning.
2. Carrying a 24,000 pound payload to GEO.
3. Carrying a full crew of 3.
4. Lasting for 14 days.

This worst case scenario may never be realized within in the first few years of deployment. One reason is that current satellites rarely weigh over 10,000 lbm. Another reason is that if the mission is simply to repair a satellite, it is highly unlikely that a crew of three will be required or that they will need 14 days to complete the mission. However, the Project Orion team has designed its OTV in anticipation of future missions. NASA is quite intent on creating other platforms in addition to the Space Station based at LEO. Our OTV will be used to realize

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this goal. It has been designed to be capable of transporting the heavy components of a platform without being unsuitable for its primary mission of satellite repair. It has been designed to support a three person crew for a duration of time that will allow them to work on assembling the platform. Other missions that may be possible because of the constraints of our worst case scenario will be manned missions to the Moon. longer duration missions (with lighter payload requirements). higher orbit missons. or missions with more personnel (this would be accomplished by adding an additional crew module).

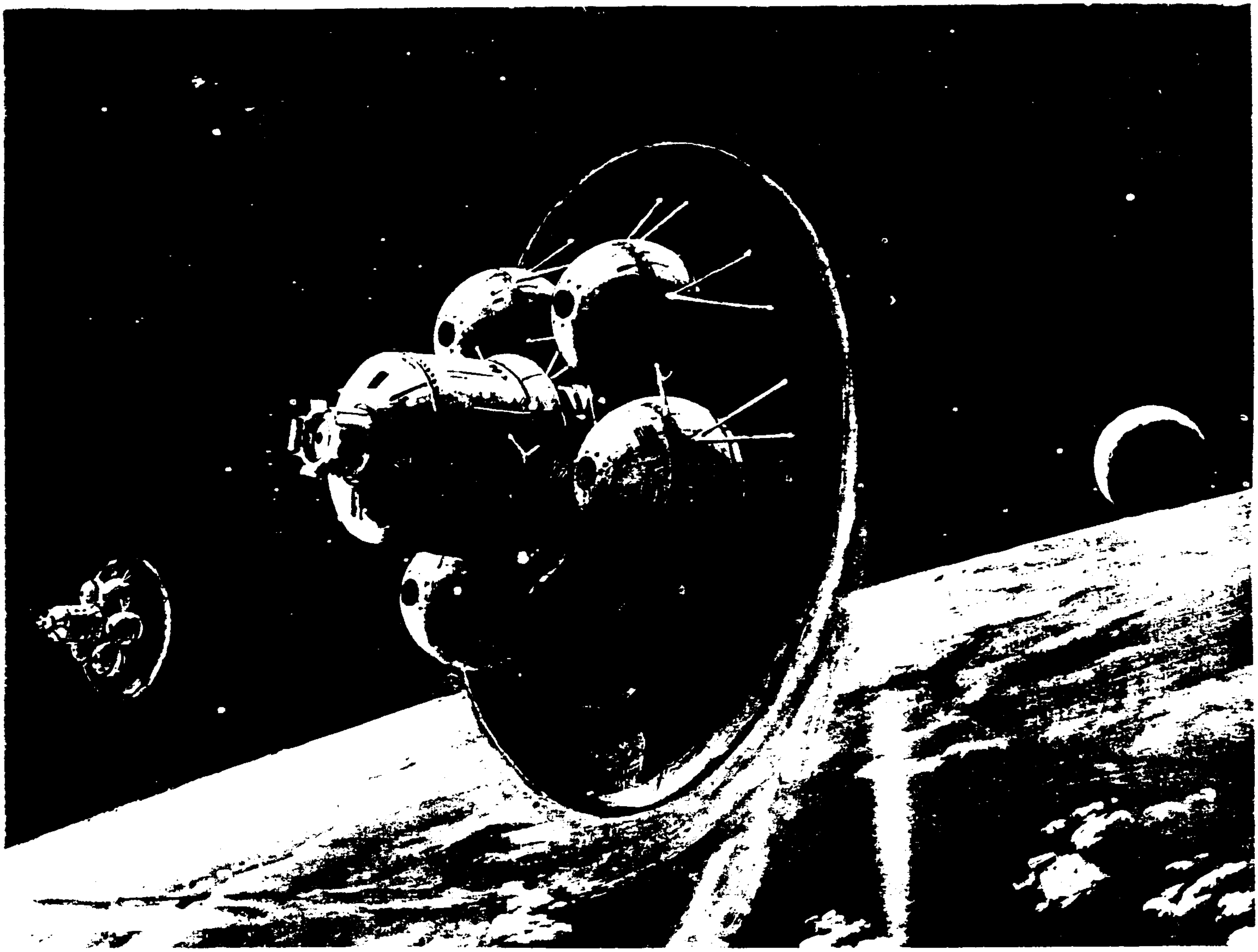

Figure I-1: An artist's rendition of an OTV similiar to WWSR's proposed design.
Source: *Pioneering the Space Frontier*

With the aforementioned criteria and worst case scenario, the Project Orion team embarked on its design process. The goal of the team is to create an OTV that will maximize performance by using the most up to date technologies. Project Orion will not use systems that have not been fully proven. It is the general philosophy of the team that it is better to stick with "tried and true" methods than risk the vehicle or the crew in order to cut costs. We also feel that using proven state-of-the-art systems will actually cut costs in the long run. One major abberation of this philosophy may be the use of an aerobrake. Early in our decision process, we selected the aerobrake as our choice for slowing down the OTV on its return to LEO. It is not a totally proven system, but it has been substantially investigated by WWSR and other companies and has shown to be highly feasible. Even so, our choice for an aerobrake is similar to the method used successfully for the Apollo missions.

The following chapters of this report consist of Project Orion's design for the OTV and its subsystems. This design has been chosen after eight months of investigation. Other designs for OTVs that use electrical, solar, or nuclear power may be more efficient. We feel, however, that our design is the most optimal possible to meet the National Commission on Space's demand for a chemically-powered, aerobraked, manned OTV as well as the design scenario selected by WWSR and MOVERS.

Chapter 1

Design of the OTV

The final design of the OTV involved the integration of many different systems. The main design parameter was the aerobrake, after that the propellant tanks, and then the cargo, and crew module areas.

The aerobrake is the most important part of the design. The the type of shield that was selected is called a raked sphere cone lifting brake. This shape was chosen so that maximum propellant savings would be obtained during the return trip to the Space Station. The brake will be shipped up tho the Space Station via the Space Shuttle. It will assembled and attached to the OTV at the station. The brake will be made of numerous sections each of which will be small enough to fit in the Space Shuttle cargo bay. The aerobrake is covered in more detail in Chapter 2 of this report.

The design of the propellant tanks was chosen with ease of construction and delivery in mind. The propellant tanks are modular and consist of two tanks (LH2 and LO2), the required support structure, and piping. The tank modules will be built on Earth and flown up empty on the Space Shuttle. The tanks will then be moved to the OTV area of the Space Station for integration to the OTV. The tank modules are designed to be identical and interchangable. The OTV has been designed to carry anywhere from two to six sets of tanks depending on the mission. These tanks are attached radially around the central command module. The tanks are put into place by cranes in the OTV servicing area. The fuel lines and diagonal supports are connected by attending astronauts. The modular design shortens the time needed for servicing the OTV, thus reducing costs.

Since the OTV has to travel in space as well as through a portion of the atmosphere, the placement and design of the crew command module (CCM), EVA module (EVAM), and cargo area are very important. The semi-spherical design of the aerobrake made it necessary to put the manned portions of the craft along the central axis. The interior components needed to positioned as symmetrically as

possible to ensure the center of gravity was near the central axis. The central location of the manned portions also means that this area will be better protected from the atmospheric heating during the aerobraking maneuver. The area protected by the aerobrake will form a cone above the brake. In order to keep the components of the OTV as well as its payload within this cone of protection meant that the central structure must be narrow but not excessively tall.

The CCM and EVAM are designed to be transported in the shuttle cargo bay. The CCM is 22 ft in length and 12 ft in diameter. The CCM contains all of the supplies, perishables, computers, controls, and facilities needed for a 14 day mission. Interior components of the CCM are broken down into hexagonal sections that fit within the circular cross section of the main pressure walls. The galley, shower, and head are in the extreme rear of the CCM. The computers and avionics are placed in front of these sections so that they are closest to the cockpit area. The life support, electrical power, and air revitalization systems are located in modules place in the "floor" and "ceiling." Unlike the rest of the CCM, the cockpit area maximizes space by returning the circular cross section. The two pilot's seats are located side by side facing forward. The controls are placed in a manner similiar to that of the Space Shuttle's cockpit. Below the cockpit is the hatch to the EVAM. The third crew member will have a seat underneath and behind the cockpit such that he would be facing the hatch to the EVAM. This seat will fold up when not in use. The area then can be used to prepare for entering the EVAM.

The EVAM is where the MMU, equipment, and tools for satellite repair will be stored. The EVAM contains an airlock that will be used to transfer between space, EVAM, and CCM. The rest of EVAM will be normally left evacuated. Outside the EVAM is the robot arm that will be used to grapple satellites and MMUs. The main EVA hatch will also double as the hard docking hatch when the OTV is at the Space Station. The EVAM can be detached from the CCM. This allows the versatility of adding any sort of module such as another crew module or space laboratory that might be needed for a given mission. The EVAM is is 8 ft in length and 10.5 ft in diameter.

The engines are placed centrally for several reasons, the most important of which is stability. The engines' center of thrust will be in line with the center of gravity of the whole OTV. The central placement will also reduce the number of lines needed from the propellant tanks and simplify servicing the OTV. Two engines acting redundantly were chosen over one main engine since this provided for a safer and more reliable system.

1-1: Detailed Drawing of WWSR's OTV

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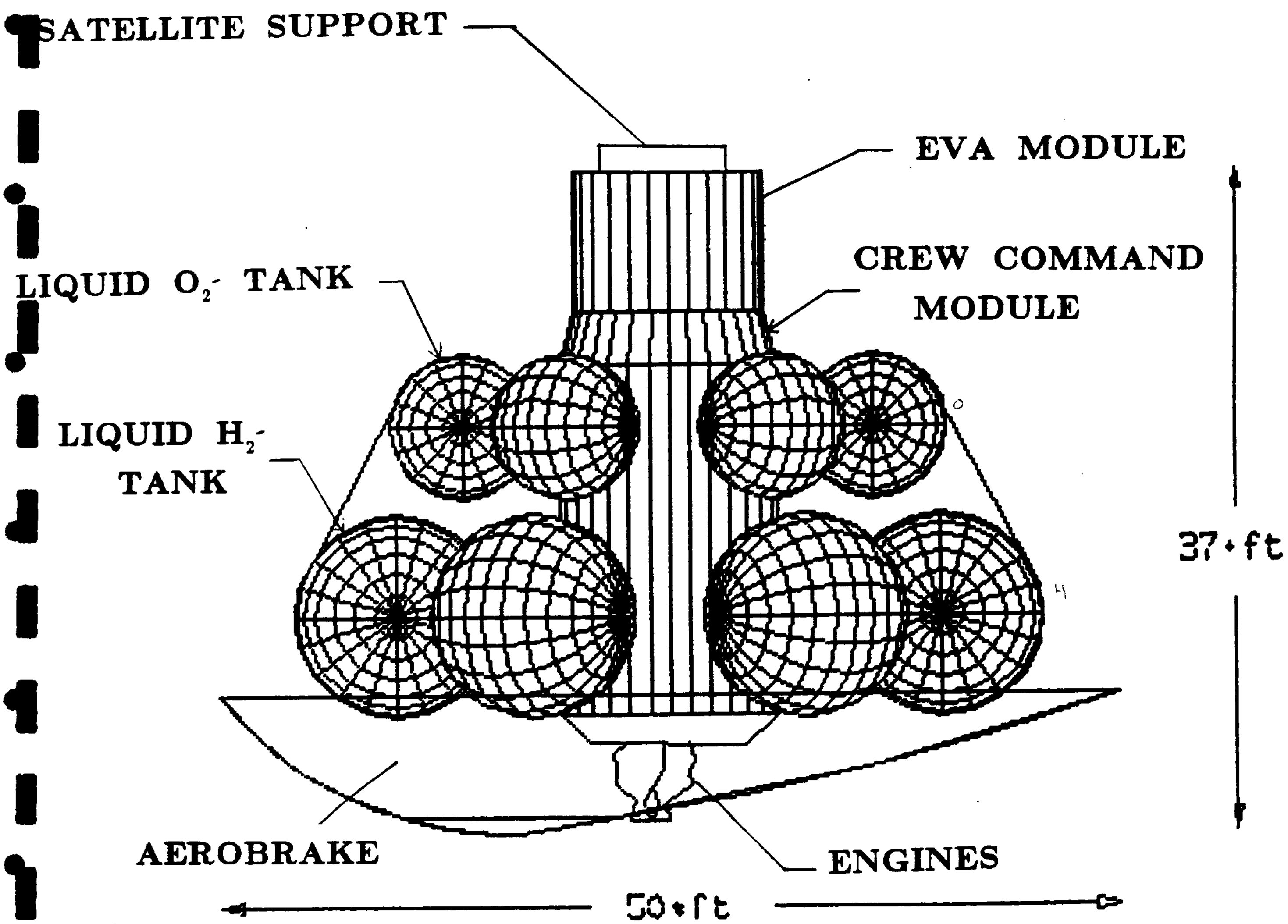
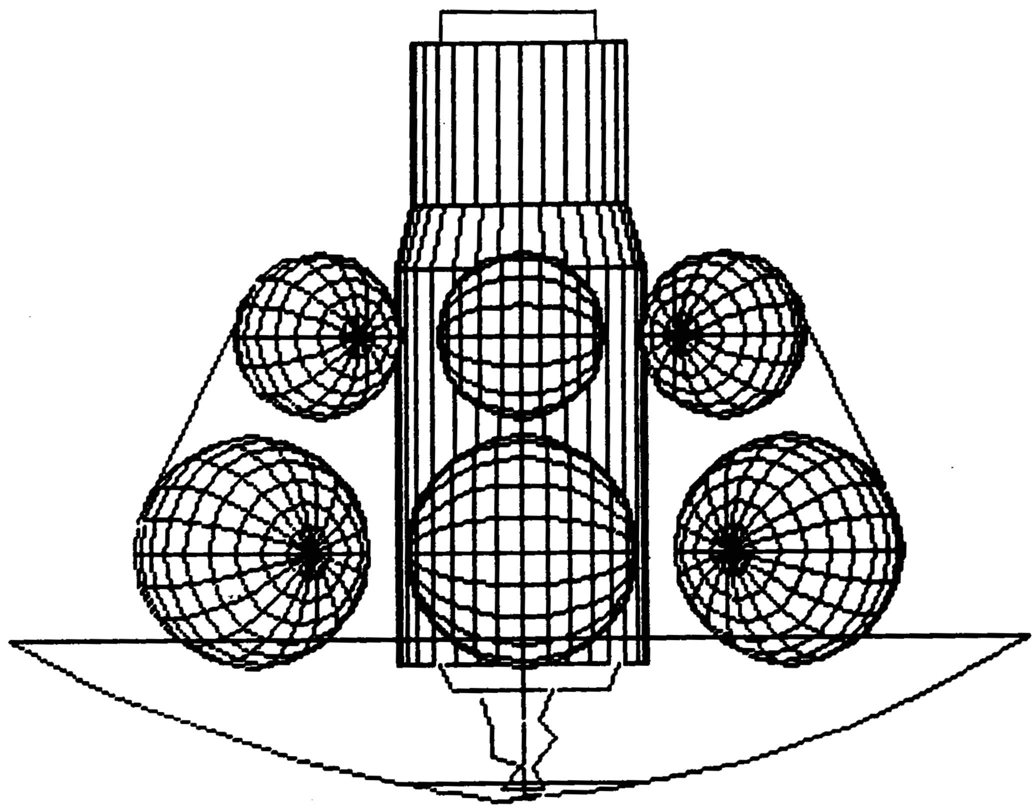
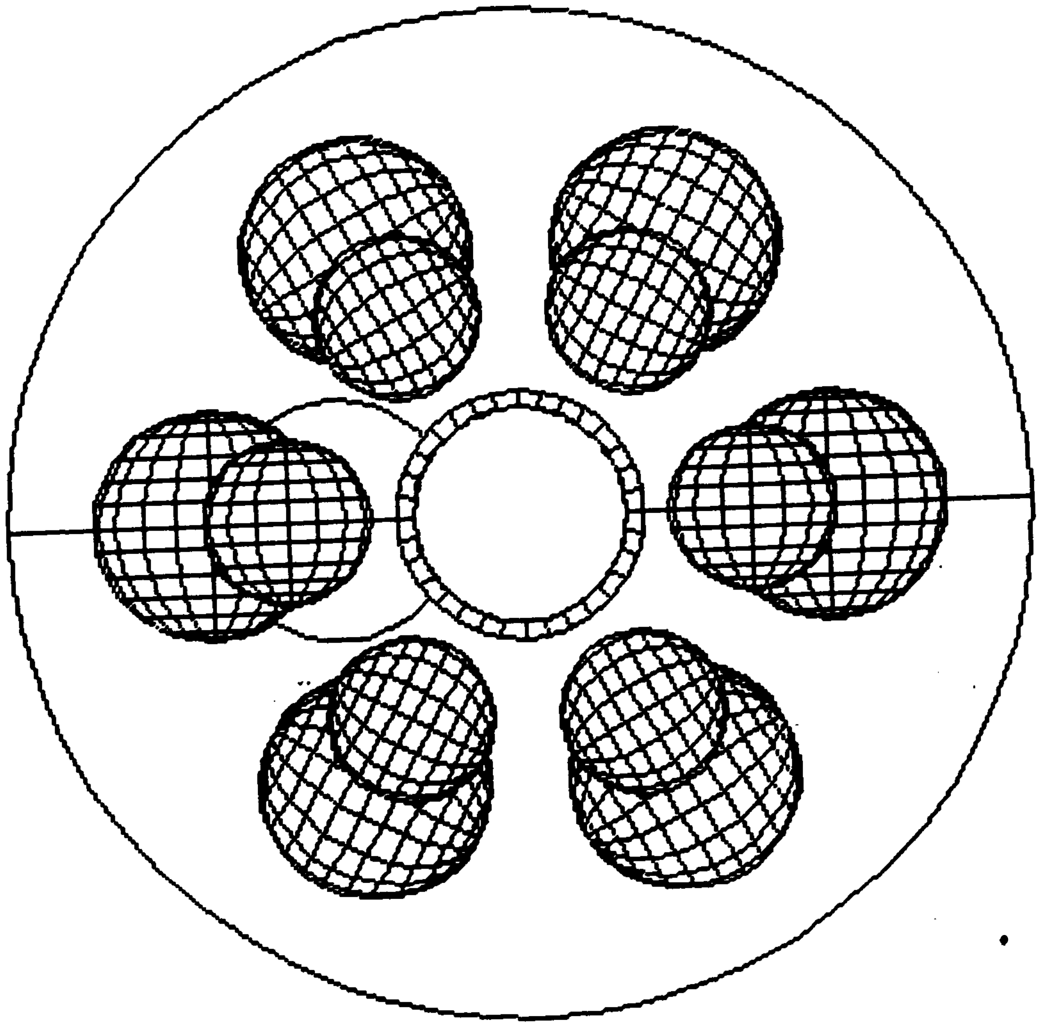


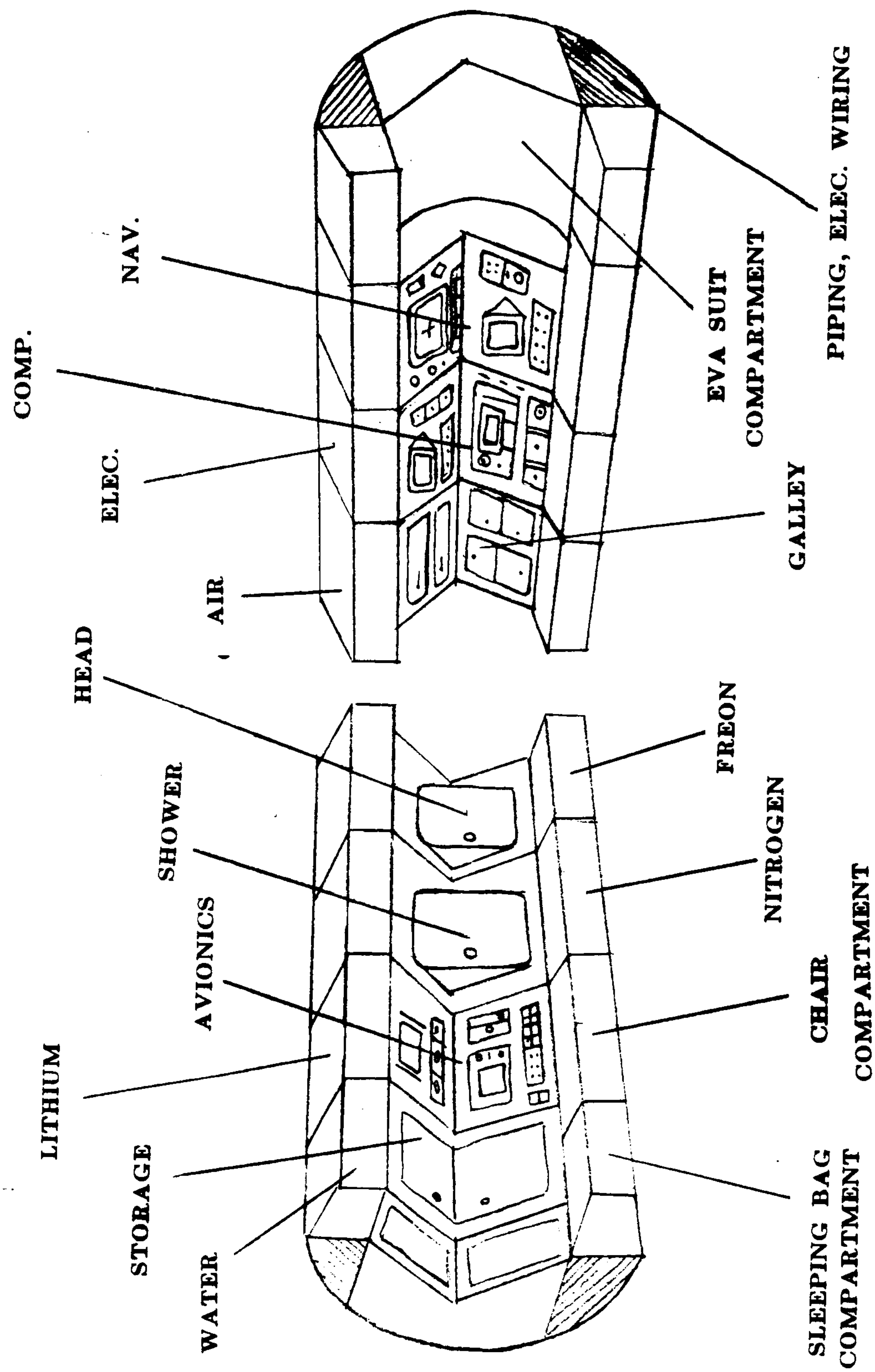
1-2: General Configuration of WWSR's OTV (Side View)



1-3: General Configuration of WWSR's OTV (Front View)



1-4: General Configuration of WWSR's OTV (Top View)



1-5: Detailed Drawing of Interior Layout

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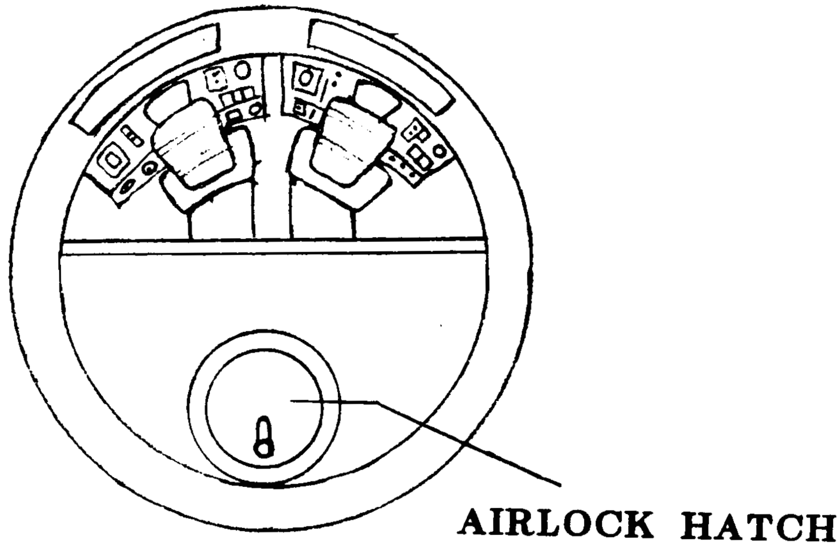
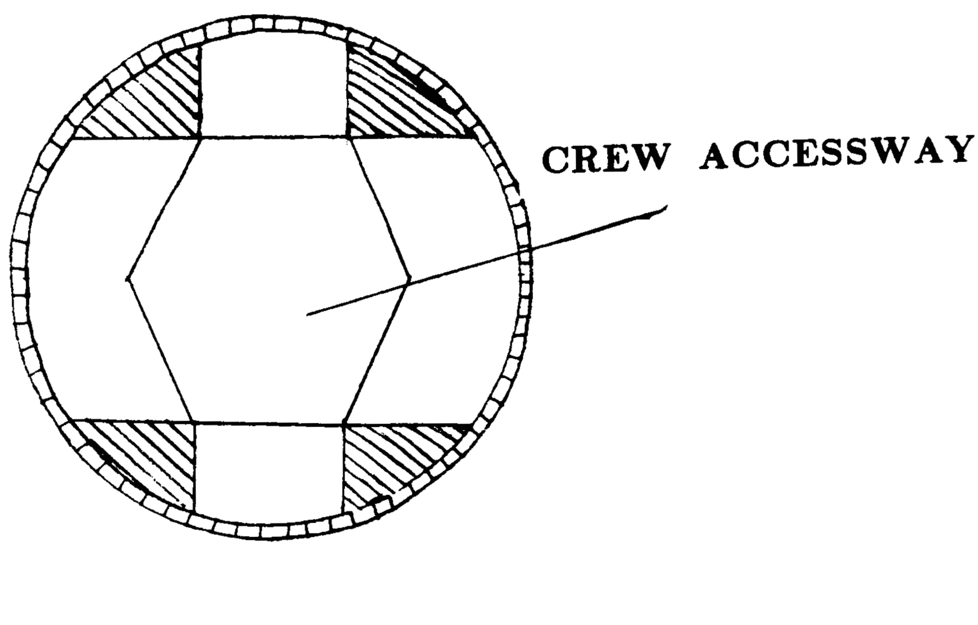
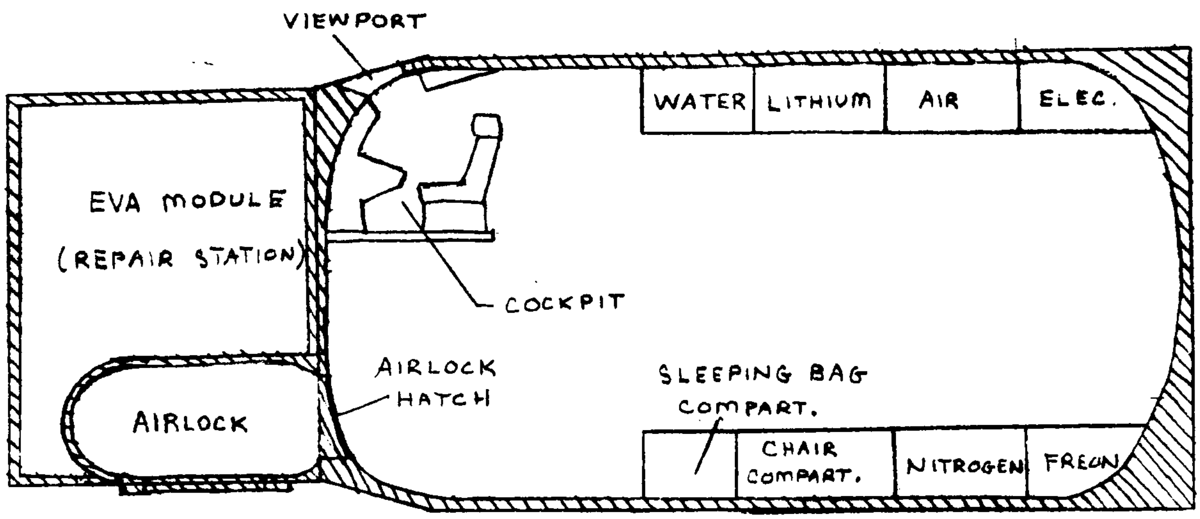


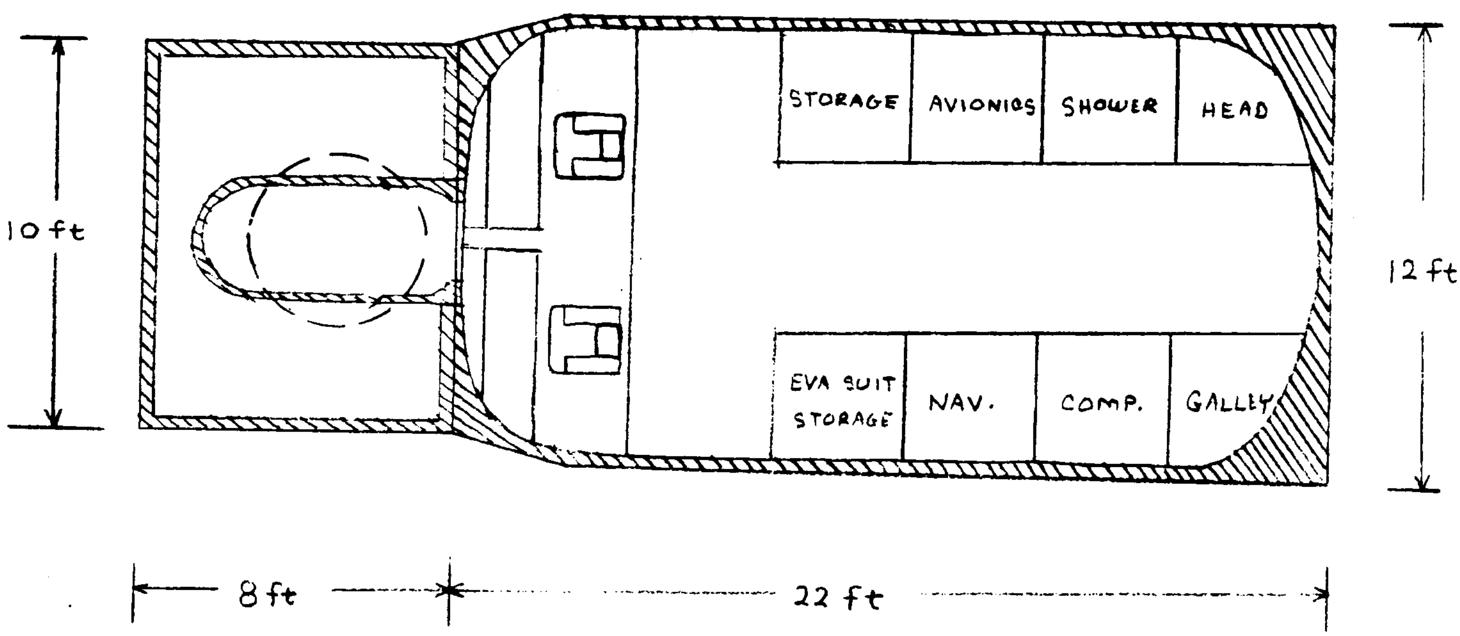
1-6: View of Cockpit as Seen from Interior



1-7: View of Interior from Cockpit



1-8: Diagram of CCM & EVAM (Side View)



1-9: Diagram of CCM & EVAM (Top View)

Chapter 2

The Aerobrake

The Aerobraking Maneuver

The WWSR OTV was designed to carry large payloads to geosynchronous orbits. In order to maximize the weight of this payload while keeping the amount of fuel needed low, the OTV will use the drag produced by passing through the Earth's atmosphere to dissipate its excess velocity as it returns from GEO. Aerobraking, as this process is called, results in a large fuel savings over propulsively slowing the craft using retrorockets. In fact, it has been shown that an aerobraking OTV can carry twice the roundtrip payload to GEO as a similiarly configured all-propulsive craft [12]. Aerobraking is, however, a very complex maneuver, creating many important vehicle design considerations. As the vehicle passes through the atmosphere it experiences severe aerodynamic heating, requiring the added complexity of a thermal protection system. Additionally, since the craft is essentially flying, aerodynamic configuration and control become prime design criteria.

The aerobraking maneuver is initiated at GEO where the OTV's engines are fired to produce the necessary plane change and inject the vehicle into a transfer trajectory that will take it into Earth's atmosphere. For most WWSR OTV missions (no returning payload), aerobraking will be performed in two passes through the atmosphere. A schematic diagram of a two pass maneuver, as compared to a one pass, is shown in Figure 2-1 [10]. The first pass will last only 5 minutes and will take the OTV to within 85 kilometers of the Earth's surface. The deceleration of the vehicle due to the drag on the aerobrake will place it in a intermediate orbit with an apogee midway between LEO and GEO. Slight corrections in this orbit will take the OTV through the atmosphere for a second time, at approximately the same altitude but for 11 minutes (due to the already reduced velocity of the OTV). This pass will place the craft in an orbit that can be circularized at LEO with a relatively small propulsive burn (less than 200 m/s delta-v).

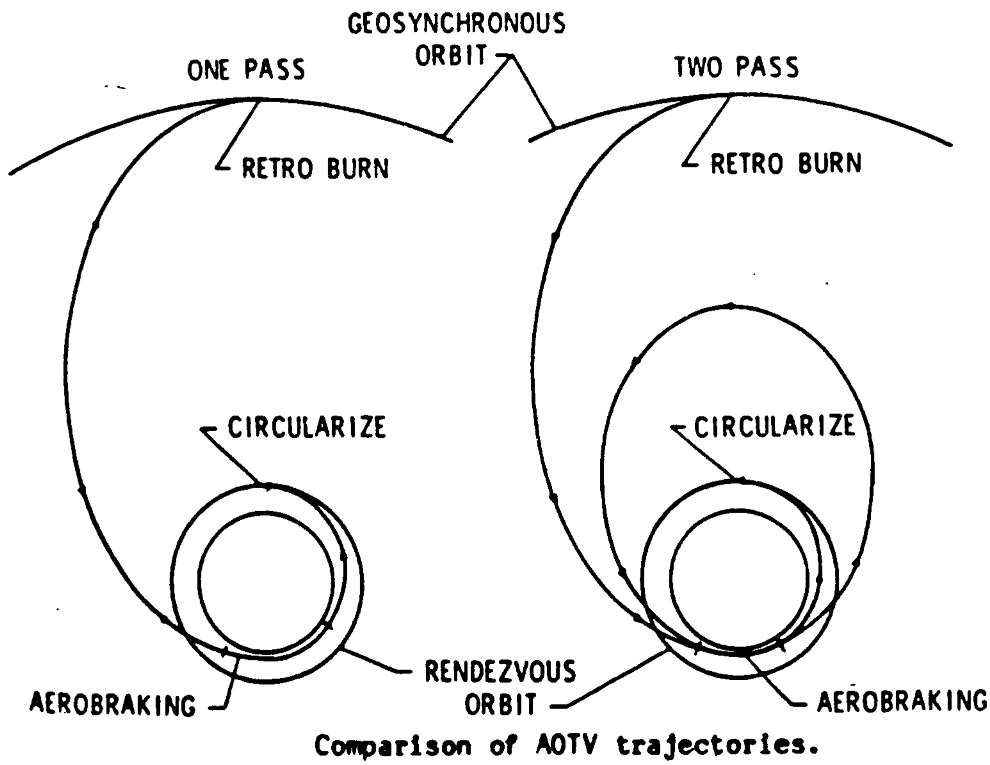


Figure 2-1: Schematic drawing of a one and two pass aerobraking maneuver.

The altitude and velocity of an unloaded (nominal return configuration) WWSR OTV versus the time into the aerobraking maneuver are shown in Figures 2-2 & 2-3, respectively. As can be seen, the first pass is a quick dip into the atmosphere that reduces the OTV's excess velocity by approximately 730 meters per second. The second pass takes the OTV down into the atmosphere almost as quickly as the first. However, because of the reduced velocity of the OTV the time for the vehicle to climb out of the atmosphere is much longer. It is during this climb out period that the major portion of the velocity decrease due to aerobraking occurs. This second pass reduces the excess velocity of the craft by 1610 m/s while producing a maximum heat transfer rate that is slightly less than that of the first pass. The second pass leaves the OTV in an orbit that can be

circularized at LEO by a small propulsive burn (200 m/s versus a burn of approximately 2400 m/s needed for an all-propulsive return to LEO). Both graphs were constructed using data obtained from a computer program used to solve the differential equations of motion of the OTV through the atmosphere [4]. The graphs shown are for an unladen OTV returning from GEO. For an OTV returning a heavy payload to low Earth orbit (LEO) the option exists to make three passes in order to keep the heating rates low.

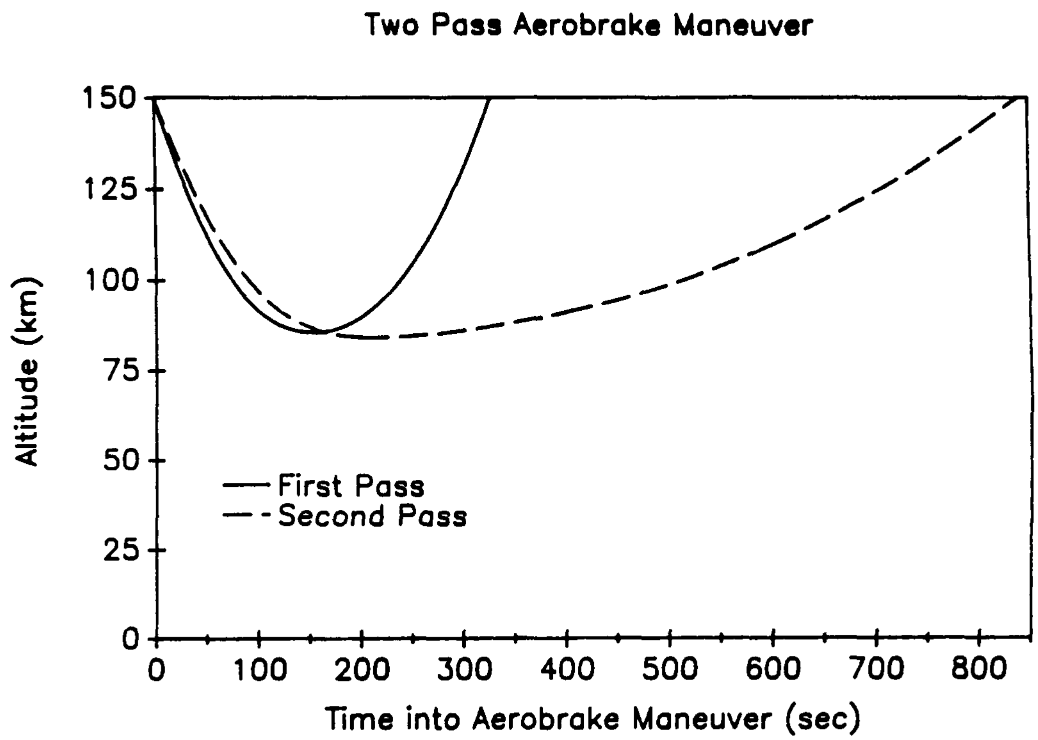


Figure 2-2: Altitude history of A WWSR OTV during aerobraking.

In order to increase the safety and lower the heating rates of the aerobraking maneuver the OTV flies through the atmosphere with a negative L/D [6]. Essentially, the vehicle is flying upside down, using the lift produced by the brake to pull the craft towards the Earth. This has two distinct advantages. If the OTV were to encounter higher than expected densities, which could catastrophically slow the vehicle sending it crashing to the Earth, the vehicle can rotate around its axis to produce a positive lift. This will increase the altitude of the OTV and reduce the deceleration. This is discussed further in the section on Aerobraking Guidance and Navigation. The second advantage of flying with a negative L/D is

that it reduces the maximum heating rate by allowing the vehicle to make longer, but shallower passes through the atmosphere. Because the vehicle remains in the atmosphere longer, it can pass through at higher altitudes to produce the same deceleration. Higher pass altitudes result in lower heating rates. This effect is discussed further in the Aerobrake Heating Section.

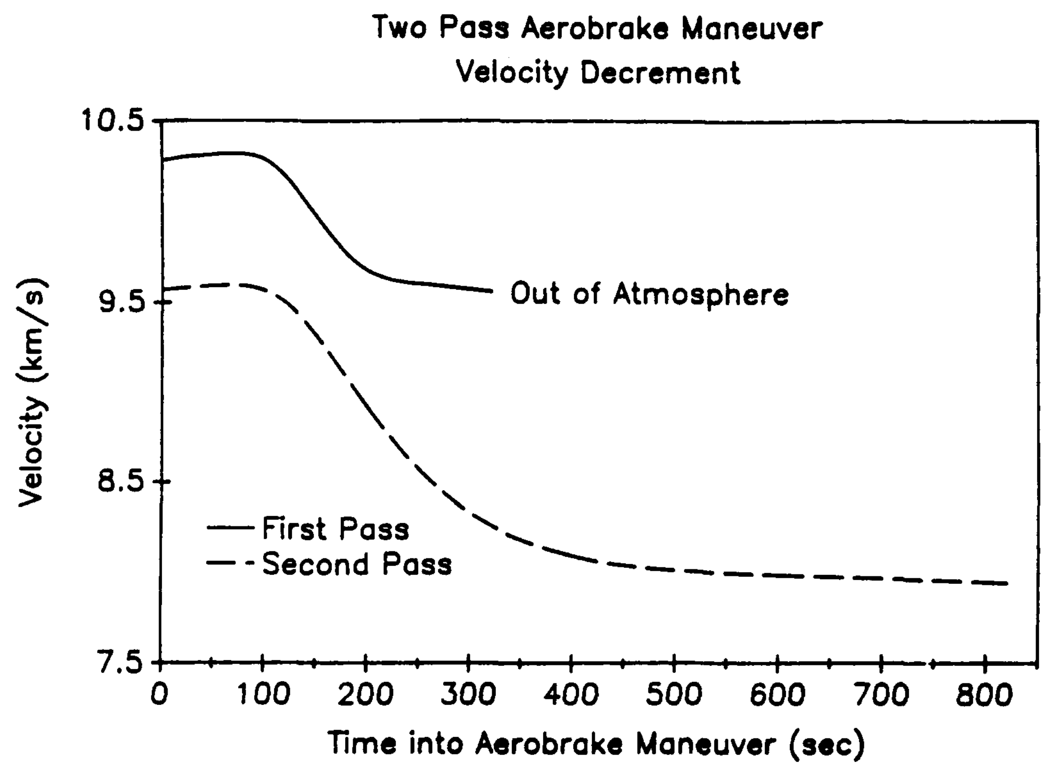


Figure 2-3: Graph showing velocity decrement of OTV during each pass of the aerobrake maneuver.

The two pass aerobrake maneuver was chosen for a number of reasons. Most importantly, it provides a margin of safety. Aerobraking the OTV in one deep pass, a maneuver called aerocapture, is possible, however, slight errors could prove disastrous. If the OTV were to encounter a higher than predicted air density on such a deep pass into the atmosphere, the velocity decrement due to drag would be so large that the vehicle may not be capable of pulling out of the atmosphere. By making two, shallower passes the effect of this type of variation can be reduced and easily counteracted.

The total aerobrake maneuver, from GEO to injection into LEO, will take only 8.6 hours; only 15 minutes of which is actual aerobraking. The maximum

deceleration due to aerobraking will be approximately 1.5 g's (for a two pass maneuver). This is below the maximum accelerations that will be encountered during other phases of the mission such as engine firings.

Aerobrake Design

The design of the aerobraking device for the proposed OTV has proven to be the basis upon which the majority of the other systems have been based. The aerobrake design affects the orbital mechanics of the OTV, the materials required, the control systems, and the treatment of heating effects. For our OTV, we have chosen a raked sphere-cone (see Figure 2-4). This design has a blunt nose configuration, similar to but not the same as the Apollo space capsule. Several factors lead to the selection of this aerobrake. The raked sphere-cone has a low ballistic coefficient ($W/C_DS = 10$ lb/ft^2) which makes it ideal for high altitude maneuvering where heating effects are small. In addition, it is flexible enough to require only a one to three pass aerobraking maneuver through the Earth's atmosphere during the return phase of the mission from GEO to a low Earth parking orbit.

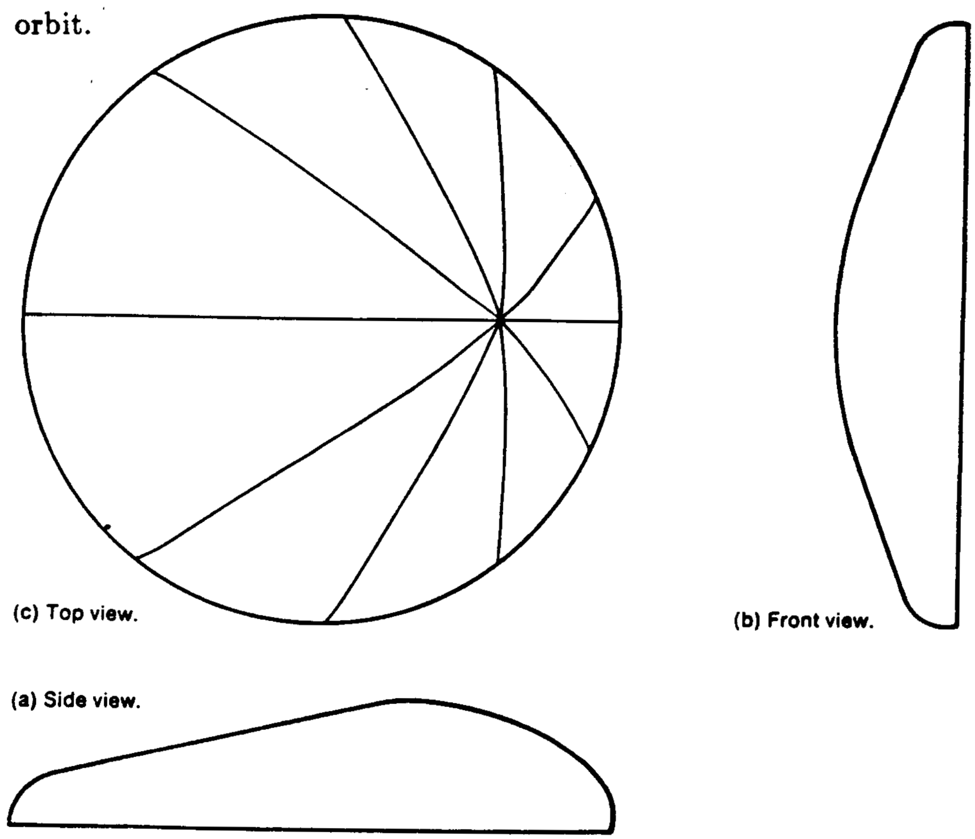


Figure 2-4: Aerobrake Geometry

We have designed our brake to be a permanent appendage to the main body of the OTV. Although the brake will only have to be removed if severe damage occurs, the aerobrake has been designed to allow for easy servicing. All servicing must be by EVA so time constraints are important. The OTV has been designed with the engines protruding through the brake in order to provide better control and stability. Some characteristics are given in Table 2-1.

Table 2-1

Characteristics of the Aerobrake

Weight	2800 lbm
L/D	0.28
C_D	2
W/C_DS	$29000/(1.6)(25)^2\pi$ $= 10$ lb/ft^2

The design of the aerobrake is derived from work by Park [9] and Bragg [1]. The fuel tanks and payload are arranged symmetrically around a reference force line (not axially). The aerobrake geometry is derived by raking-off a circular cone, blunting the apex with a spherical cap, and faring the frustrum by a fourth-order polynomial. This is shown in Figure 2-5. L/D equals 0.28, when the vehicle flies at an angle of attack of -5°, with respect to the cone axis. By shifting the LO2 from tank to tank, the c.g. can be shifted in the yaw and pitch plane, changing the angle of attack. Using this control the vehicle can even remain stable after the loss of one engine [9]. The engines have extendable nozzles that are stored flush with the heat shield during atmospheric flight. The cut-out openings for the engines are at an off-stagnation point location where the heat- transfer rates are lower. In the back side of the aerobrake, the tanks and payload are covered by a shroud which provides protection from solar flux, the heat of aerobraking, and the impacts of meteoroids and space debris.

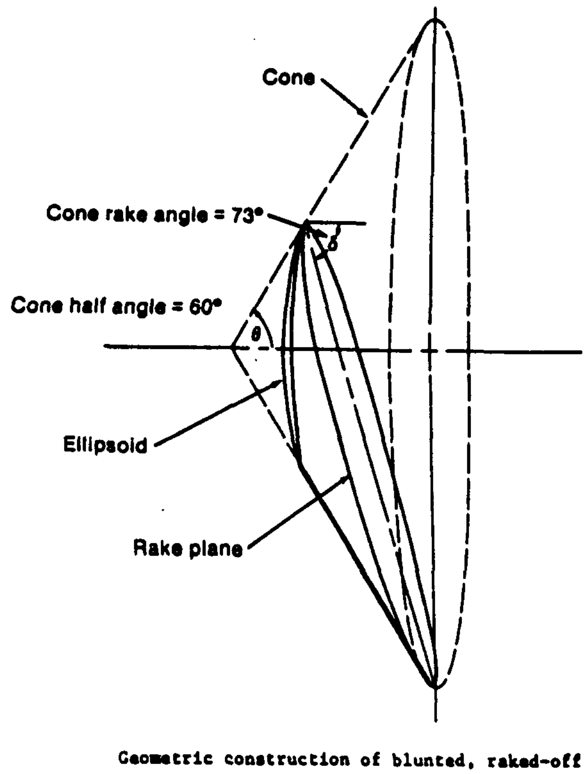


Figure 2-5: Geometric Construction of Raked-Cone

The reference force line (the X-axis, the axis of symmetry) represents the net aerodynamic force vector originating from the center of pressure at the desired flight angle of attack. The relation between the X-axis and and direction of travel is seen in Figure 2-6. As long as the c.g. moves along the X-axis, the trim angle of attack will not be affected. This means that changes in cargo and fuel loadings do not affect the trim in this design. Also, the c.g. can easily be shifted to bring the aerobrake to any desired trim angle. The c.g. can be shifted by moving LH_2 and LO_2, and also by gimballing the engines. This alone can control the navigation of the OTV or the RCS rockets can also be used to roll the entire vehicle, achieving a time average angle of attack.

The aerobrake is to be constructed of an inflexible heat shield material, cemented on metallic panels. These panels are supported by a system of beams and struts, as seen in Figure 2-7. Weight is distributed over the aerobrake, and

the structure of the aerobrake is integrated into that of the entire vehicle, thereby minimizing the total structural weight.

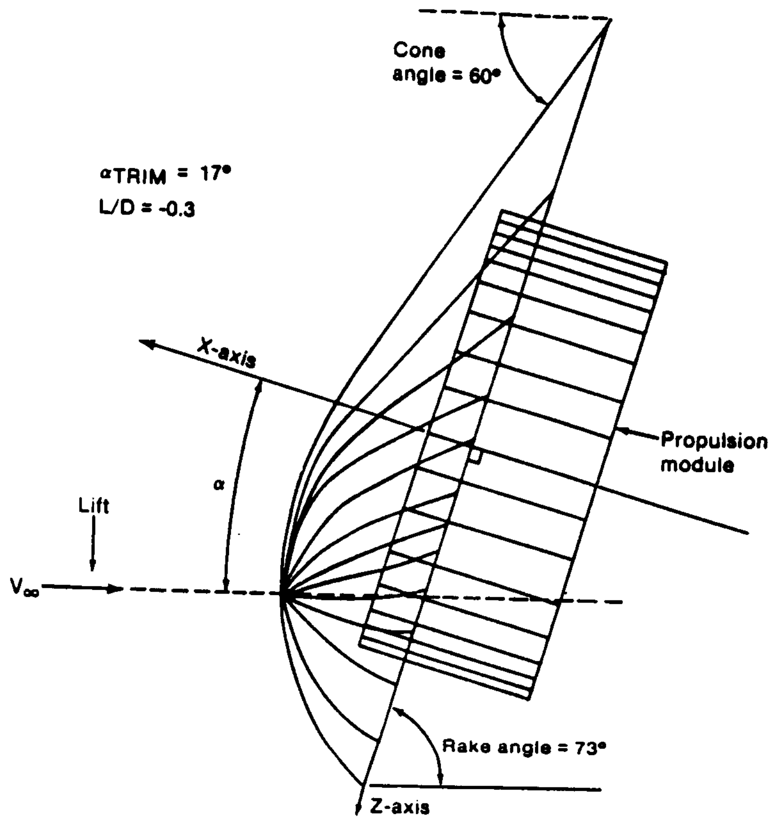


Figure 2-6: Flight Path Angle

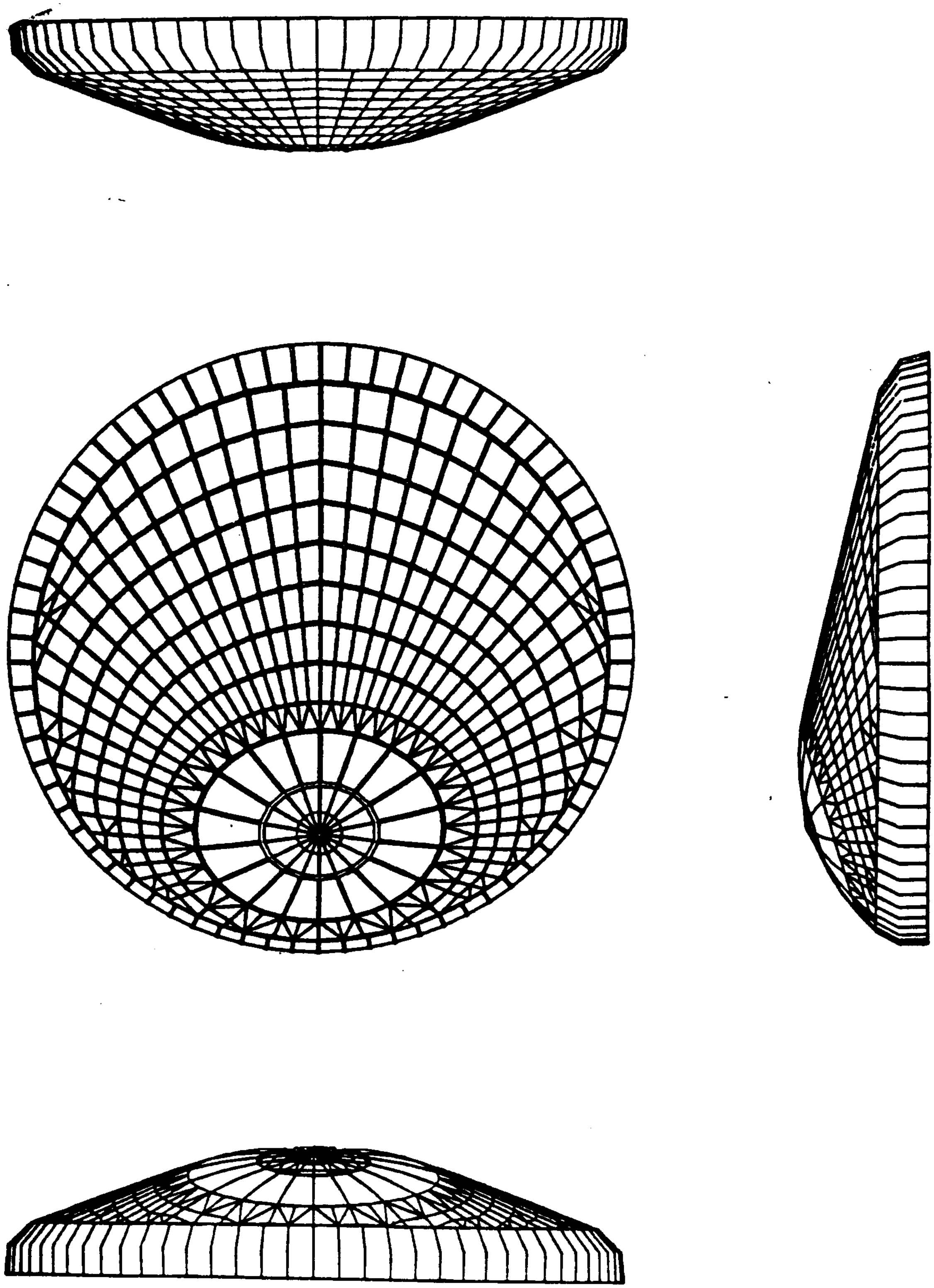


Figure 2-7: NASTRAN Model

The skeletal construction of the heat shield is shown in Figure 2-8. The structure consists of aircraft-type skin, stringer, rib, and frame construction. The skin, which serves as the inner mold line of the thermal protection system, is riveted onto the structure. In section A-A (see Figure 2-8b) notice that the ring is rolled into a circular shape. This ring has a flange for the purpose of riveting an annular closeout plate at the bottom. The peripheral bulkheads are riveted onto the ring as shown. These bulkheads have a flange to which the wraparound edge panels are to be riveted as shown in section E-E (see Figure 2-8c). The ring, annular plate skin, and bulkheads form enclosed structural boxes around the periphery of the heat shield. In effect, this provides a stiff outer hollow ring that is stiffened every 5°. This structural ring then serves to support brackets to attach the heat shield to the OTV.

The aerobrake must be transported by parts and assembled in space, because it is too large to fit in the shuttle or the aft- compartment of the external tank.

Aerobrake Heating

One of the most problematic aspects of the aerobraking maneuver is the heating of the aerobrake due to drag as it passes through the Earth's atmosphere. There are two methods of reducing the maximum heat transfer rate of the aerobraking maneuver; making multiple passes through the atmosphere and flying at a negative lift-to-drag ratio.

Multiple passes allow the OTV to make shallower dives into the atmosphere. The heating rate of the brake is reduced because the aerodynamic slowing of the OTV is performed gradually over a greater time period. Figure 2-9 shows the heating rates of a two pass maneuver relative to that produced by single pass aerobraking [4]. Making two passes results in a decrease of the maximum heating rate by as much as 30%. As seen in Figure 2-10, the two pass maneuver can be optimized to give a minimum mission heating rate. As it turns out optimization results in both passes being of approximately the same depth into the atmosphere. Minimizing the heating rates results in a slightly greater than minimum deceleration on the second pass, however, this deceleration is well within the structural and physiological limits of the OTV and crew.

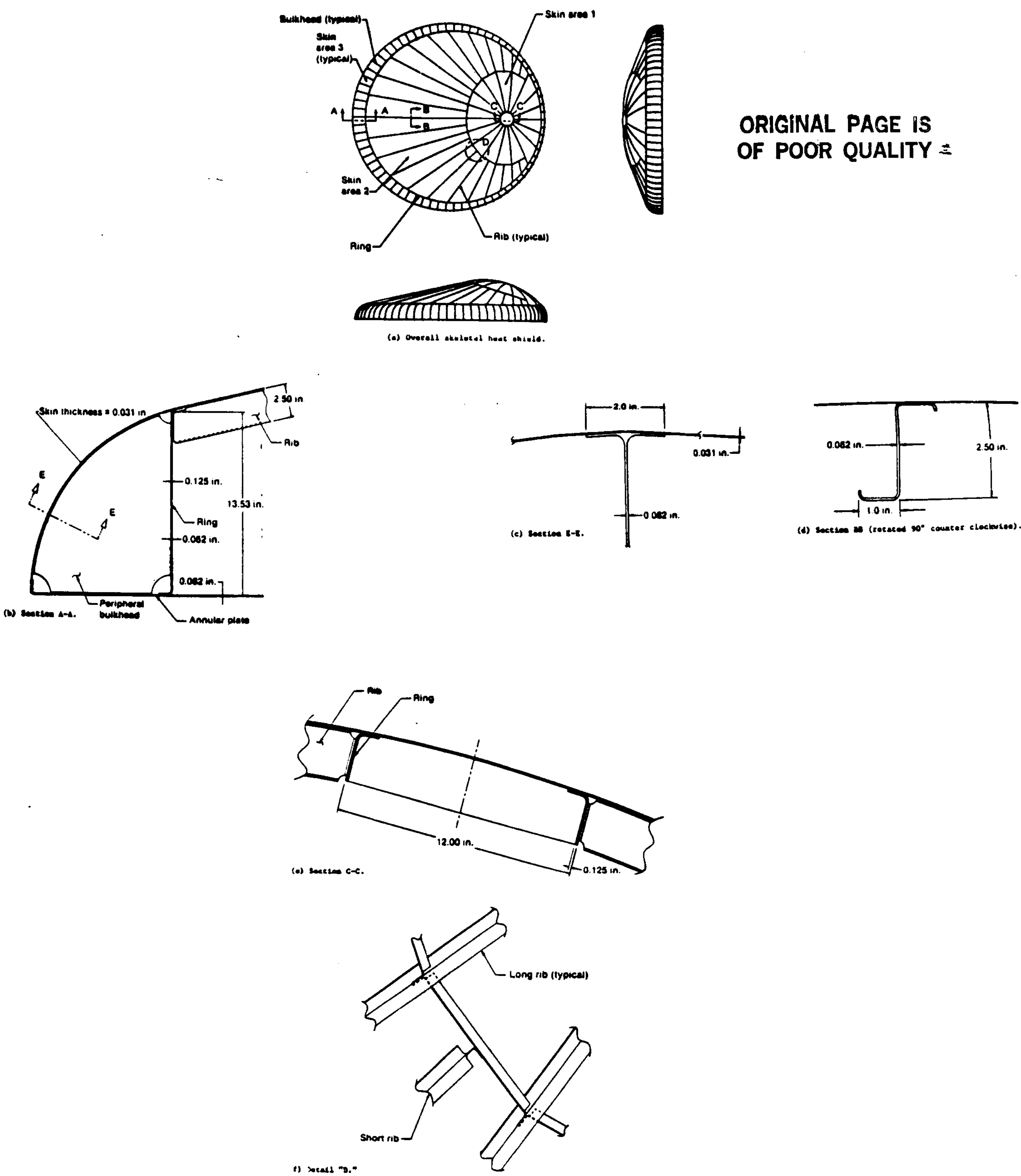


Figure 2-8: Construction of Heat Sheild

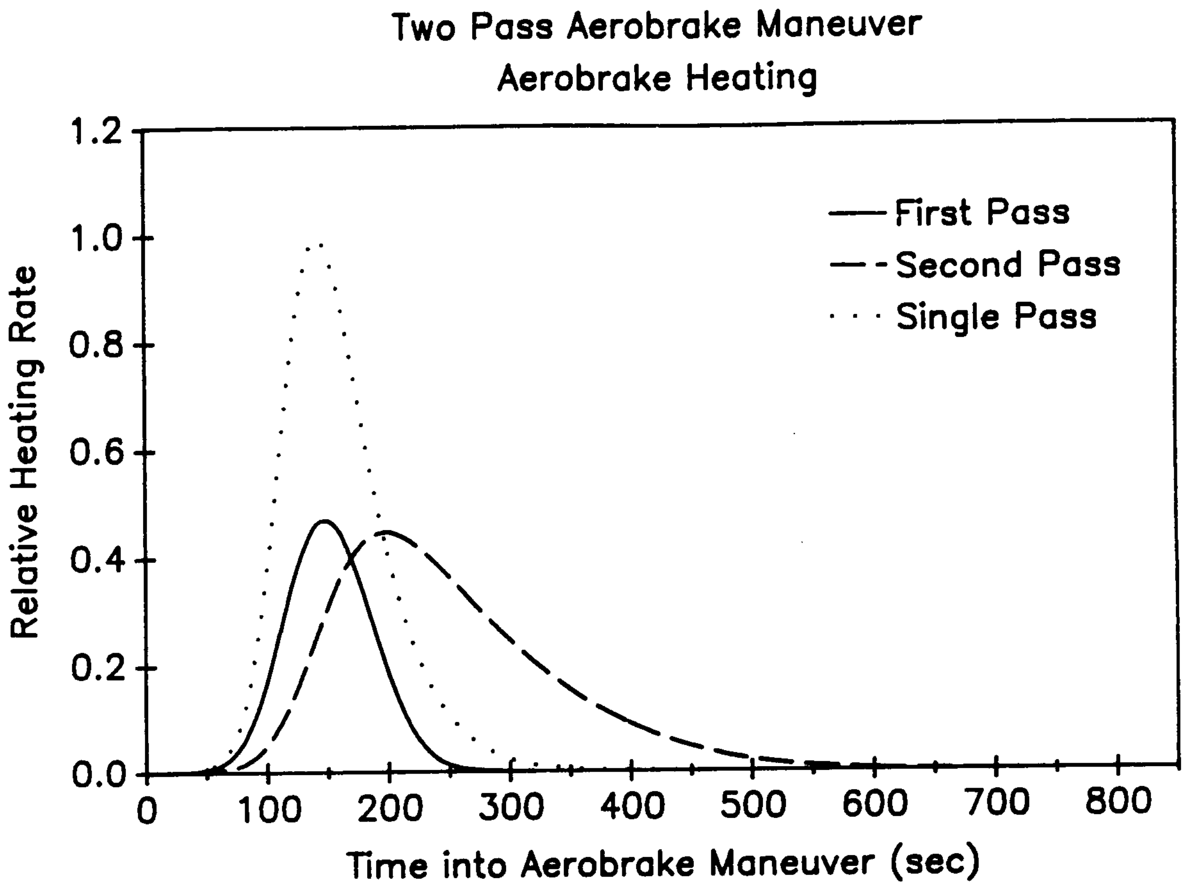


Figure 2-9: Graph showing the relative heating rates of a two pass aerobraking maneuver compared to that of a single pass.

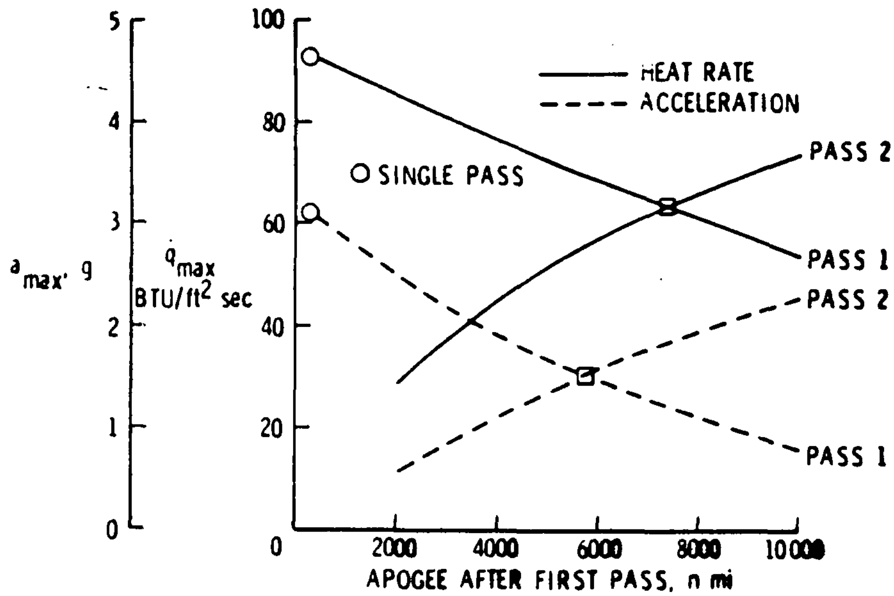


Figure 2-10: Graph showing minimization of a two pass aerobraking maneuver. Note optimization of heating rates results in higher than optimal decelerations.

Figure 2-11 shows the heating rate during aerobraking for an OTV of similar configuration as the WWSR OTV. This graph was constructed for an OTV with a ballistic coeffiecent of 11.9 lb/ft^2 making a one pass aerobraking maneuver [1]. The WWSR OTV has a slightly lower ballistic coefficient (10 lb/ft^2) and will therefore encounter lower heating rates than shown. Figure 2-12 shows this effect of the ballistic coefficient on the heating rate. Additionally, the WWSR OTV will be performing a two pass maneuver that will reduce these rates by approximately 30%. A conservative estimate of the total (convective and radiative) maximum heating rates encoutered by the WWSR OTV, as compiled from numerous sources [1,8,9,10], has been calculated as 25 Btu/ft^2-sec (28 W/cm^2). Compared with other braking configurations, such as a lifting body or aerobraking tug, the WWSR OTV will produce relatively low heating rates. The relationship of these rates to the thermal protection systems of the OTV will be discussed in a following section.

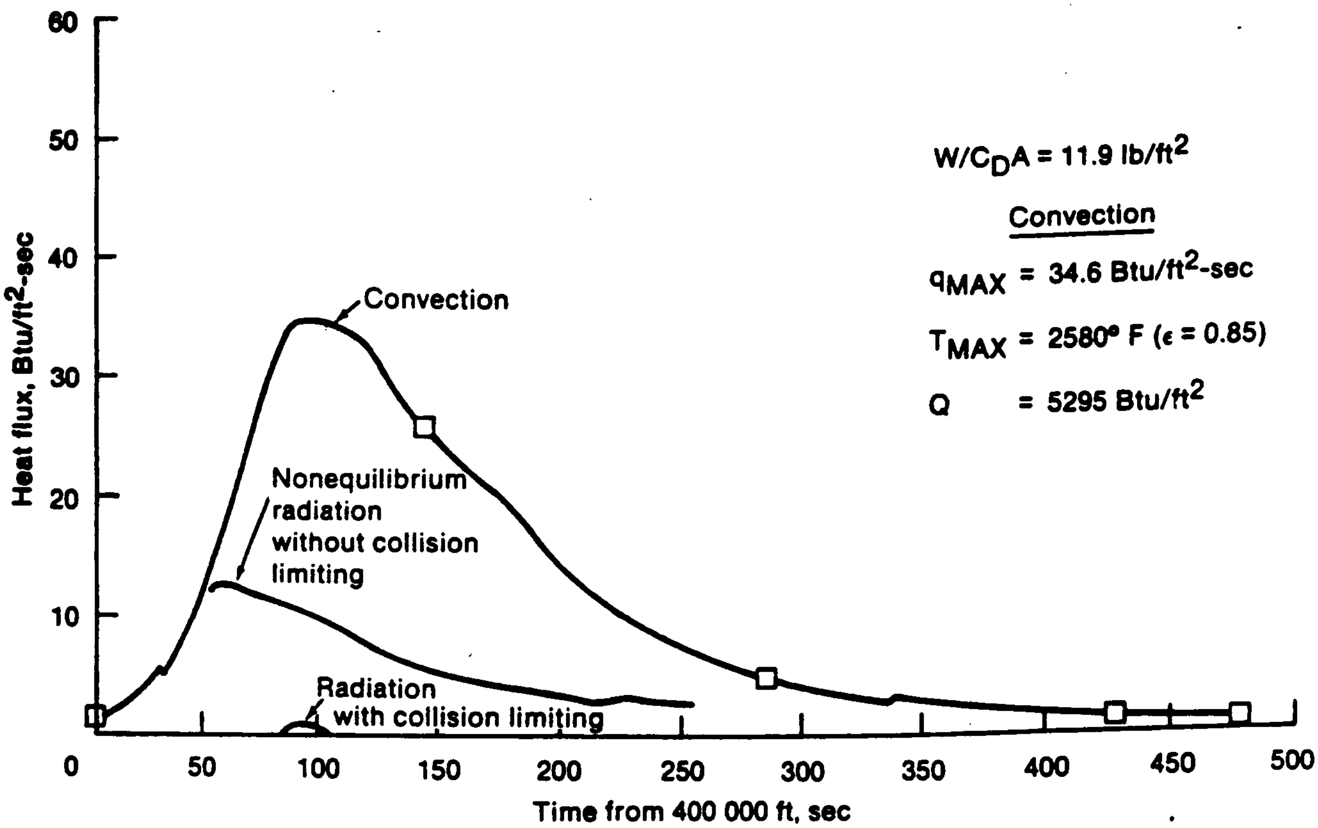


Figure 2-11: Graph showing the heating rate history of an aerobraking OTV of similiar configuration to the WWSR OTV.

In order to reduce the heating rates further, the WWSR OTV will fly through the atmosphere with its lift vector pointing towards the Earth. This allows the OTV to make a shallower pass into the atmosphere because the lift produced by the vehicle will hold it down in the atmosphere longer producing the necessary deceleration. This longer but shallower pass produces the same deceleration as a quick, deep pass but with much lower heating rates since the densities encountered in the long, shallow pass are lower.

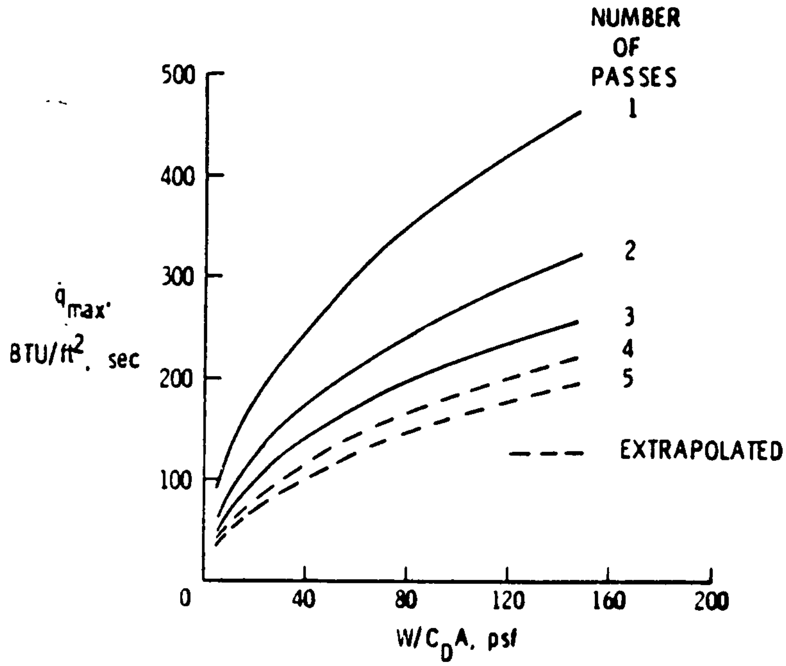


Figure 2-12: Graph showing the effect of vehicle ballistic coefficient and number of atmospheric passes on maximum heating rates.

Because of the large size of the WWSR OTV's aerobrake and the use of a multiple pass, negative lift aerobraking maneuver, the heating rates produced will be well within the limits of the aerobrake's heat shield and thermal protection system.

<u>Thermal Protection System</u>

Several studies [1,8,9,10] have shown that a one pass return trip from GEO to LEO, the raked sphere-cone with a ballistic coefficient of 15 lb/ft^2 will experience between a 35 and 40 w/cm^2 heating rate and 2 g's of decceleration. Our computer simulation and other studies [8,10] have shown that thermal and mechanical stress can be reduced by 50% for a three pass return with a negative lift vector and 30% for a two pass return with a negative lift vector. This is seen in Figure 2-13. For a one pass mission the thermal protection system (TPS) would weigh 2300 lb and the supporting structure would weigh 2000 lb for a total aerobrake weight of 4300 lb. A two pass mission with negative lift effectively

reduces the total weight of the aerobrake to less than 2800 lb. Our aerobrake is designed for 2800 lb to give a large safety margin. Menees has shown that the time required for the return trip from GEO to LEO is 6 hours for one pass and 14 hours for a three pass mission [8]. This time difference is insignificant for a 14 day mission; therefore, a multiple pass return is advantageous for weight savings and heat reduction.

As noted in the preceeding section, 28 W/cm^2 is the maximum heating rate encountered. This heating rate at only one location on the aerobrake and for only a few seconds of the re-entry maneuver. Figure 2-14 shows the drop in heat flux and pressure across the aerobrake. Notice that the heating rate is small across most of the aerobrake.

The heart of the thermal protection system is the high-temperature reusable surface insulation (HRSI) such as that used on the Space Shuttle. A cut-a-way view of the HRSI is shown in Figure 2-15. This material is a 12 lb/ft^3 fibrous refractory composite insulation (FRCI-12) consisting of sintered silica fibers reinforced with silicon carbide fibers. The exposed surfaces of the tiles are coated with reaction-cured borosilicate glass with SiB4 included as an emittance agent [1,2]. The tiles are bonded with a 0.0075 inch thick layer of RTV-560 adhesive to a 0.16 inch thick strain isolation pad (SIP) made of felted aromatic polyamide fibers (NOMEX) which is bonded to the aluminum skin with RTV-560. The thickness of the FRCI-12 is designed to limit the temperature of the outer bondline to 550 °F and the temperature of the inner bond line to 350 °F. The thickness and density of each material is given in Table 2-2.

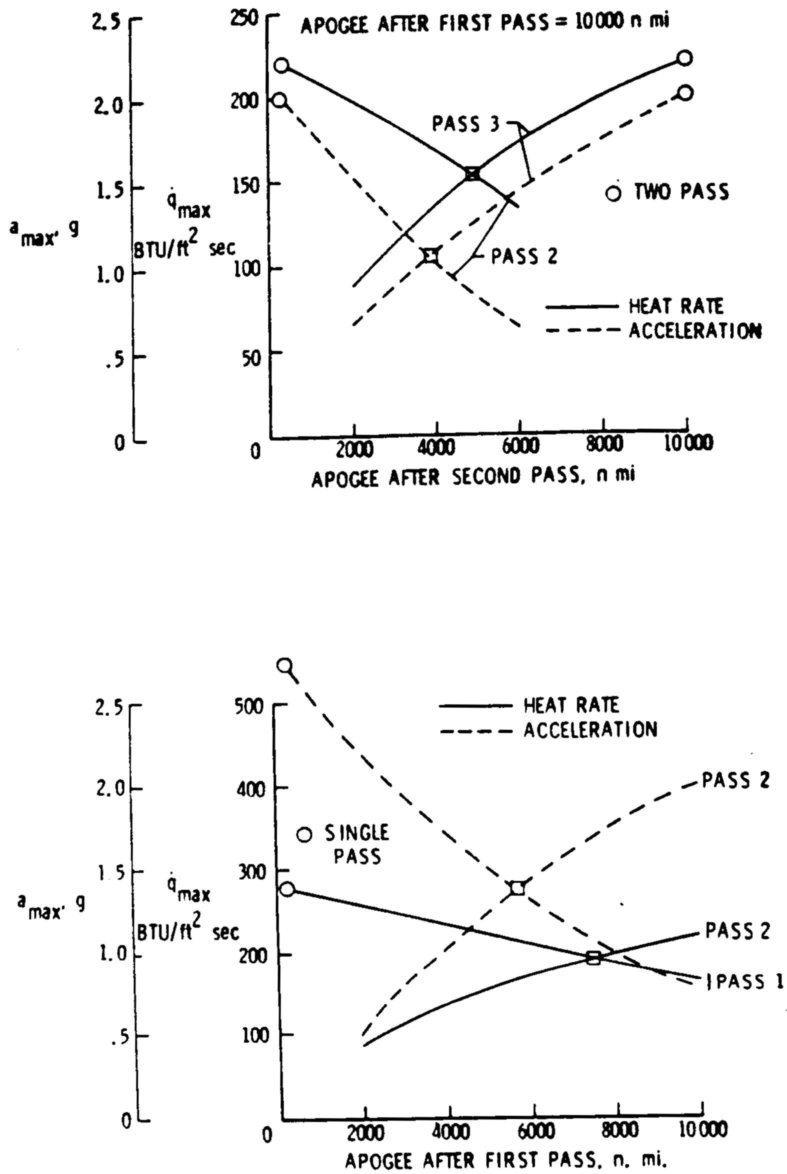


Figure 2-13: Reduction in heating rate and deceleration due to multiple passes.

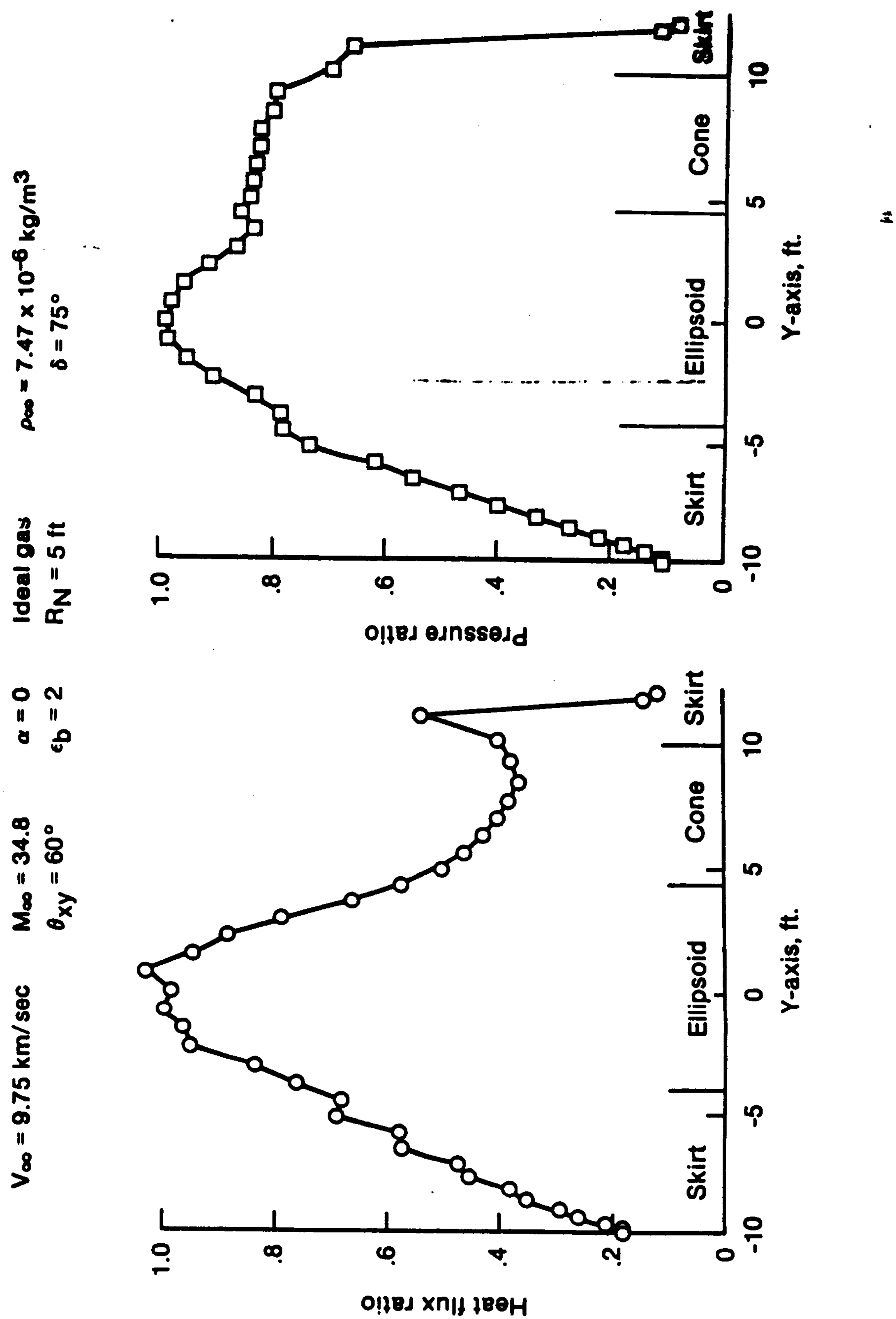


Figure 2-14: Heat Flux and Pressure Distribution

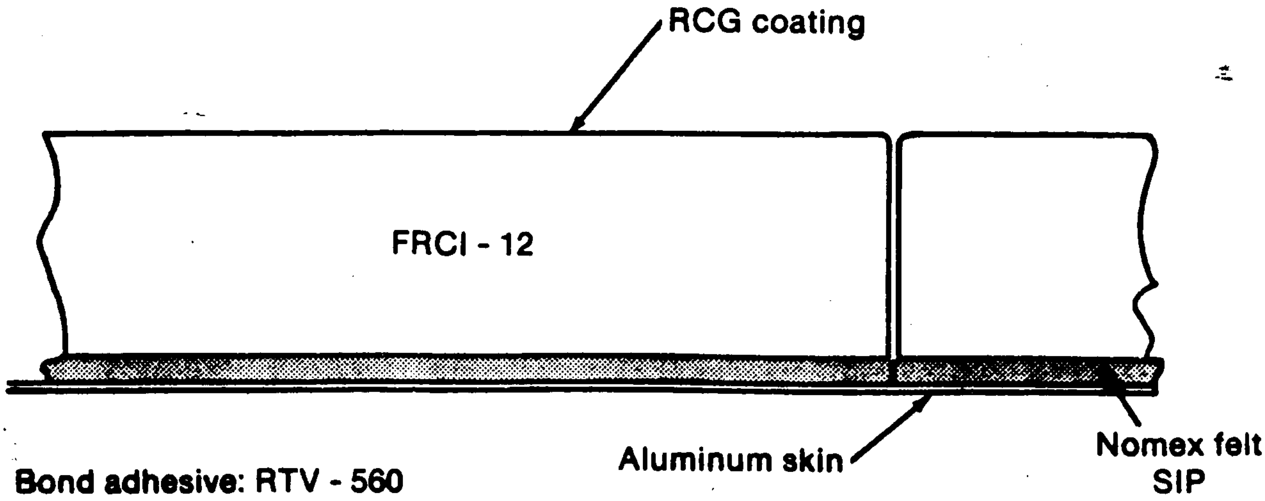


Figure 2-15: Schematic Diagram of the Thermal Protection System

Table 2-2

Thickness and Density of TPS

Material	Thickness (inch)	Density (lb/ft^3)
Tile coating	0.015	0.13
RTV-550 cement (2 layers)	0.0035 each	0.11
SIP	0.08	0.072
FRCI-12 tile	0.43	0.873

The skirt of the aerobrake, which is a region of high curvature, is covered by an array of rectangular tiles arranged in four circumferential rings. This is shown in the side view in Figure 2-16. The large, shallow cone area and the ellipsoidal nose area of the aerobrake is covered with an array of hexagonally shaped tiles (see Figure 2-16). The hexagonal shape of these tiles has several advantages over rectangular tiles. The hexagon has a smaller perimeter-to-area ratio than a rectangular or square, which results in fewer or shorter gaps between tiles. Also,

there are no long running gaps that tend to augment tile edge heat flux. Gaps are provided between tiles to accommodate the difference in thermal expansion between the tiles and the aluminum substrate, and thus prevent tile-to-tile contact. Tile-to-tile gap fillers of woven ceramic cloth are used in regions of high entry-surface-pressure gradient to prevent high tile-gap heating. The gap filler fabric is shown in Figure 2-17.

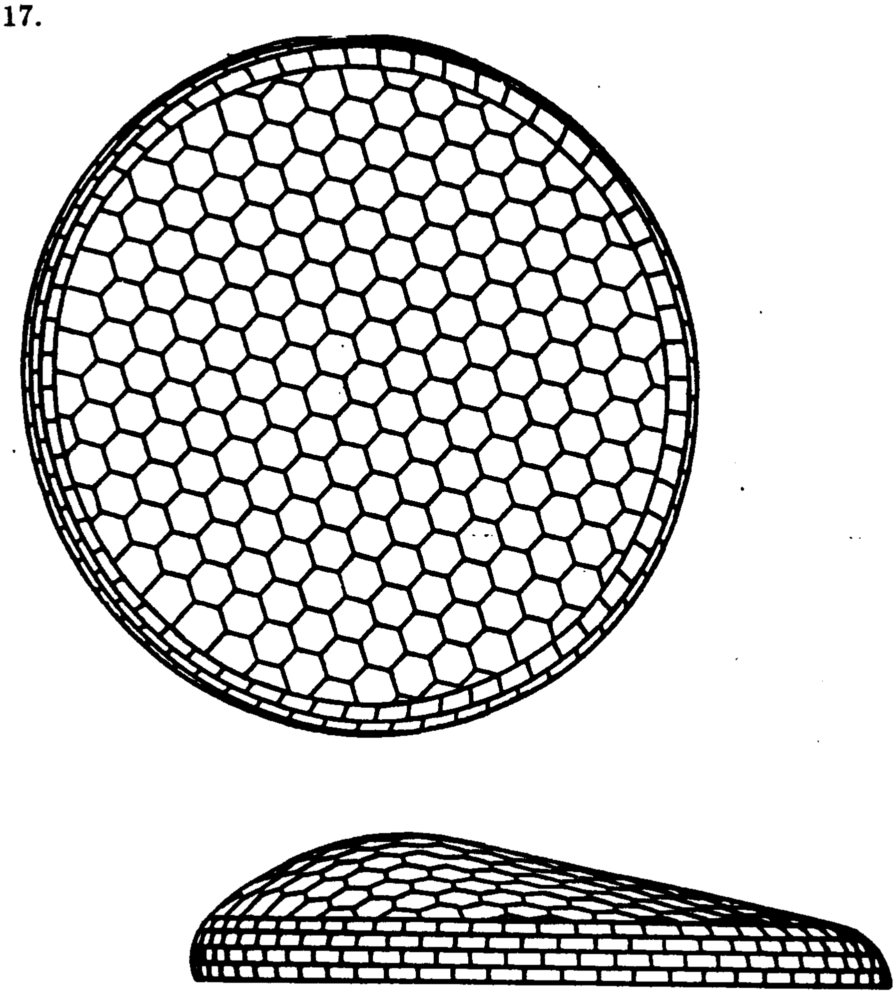


Figure 2-16: Thermal Protection System on Aerobrake

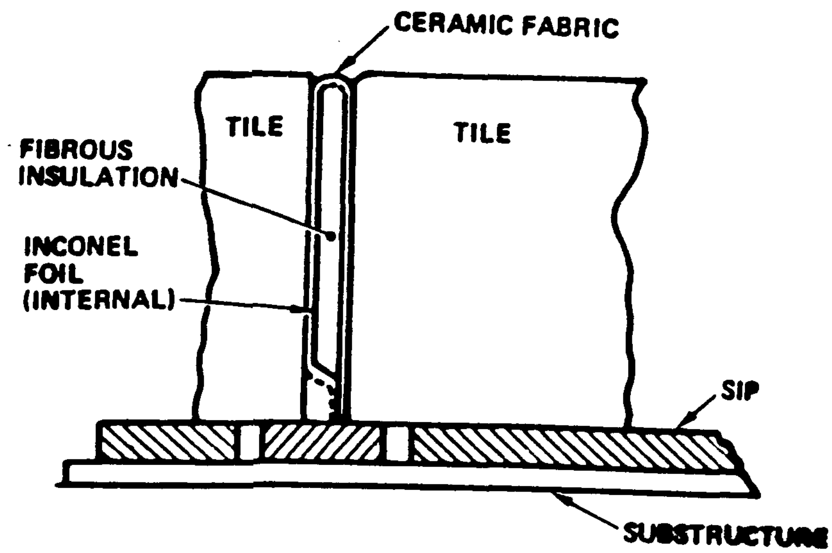


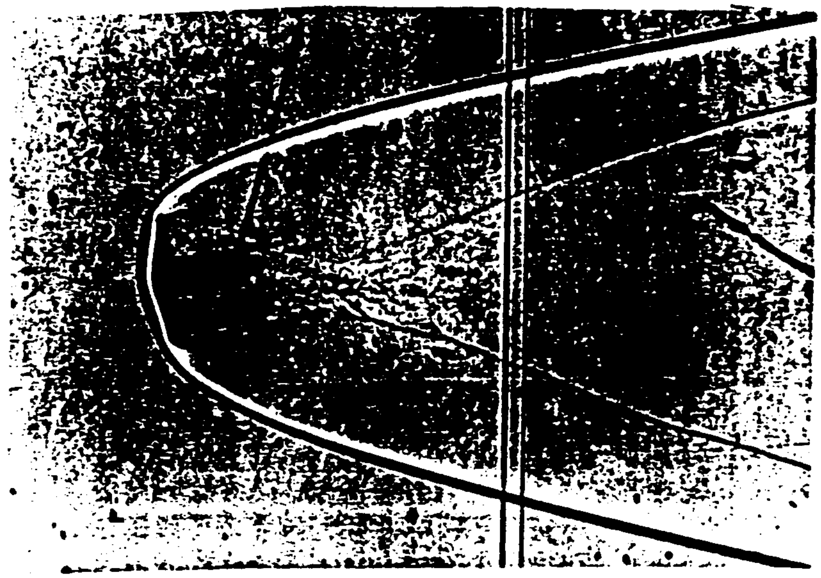
Figure 2-17: Gap Filler Configuration

The hexagonal shape also results in reduced stresses in the tile, in the tile coating, and at the tile bondline. To decrease the cost of the tiles, fewer and larger tiles are assumed rather then many small tiles.

Often a problem exists about the convective heat-transfer rates at the frustrum edge. A circular frustrum produces high convective heat-transfer rates [11]. Such occurrences of high heat-transfer rates are avoided by contouring the frustrum such that the surface curvature increases gradually toward the edge [5].

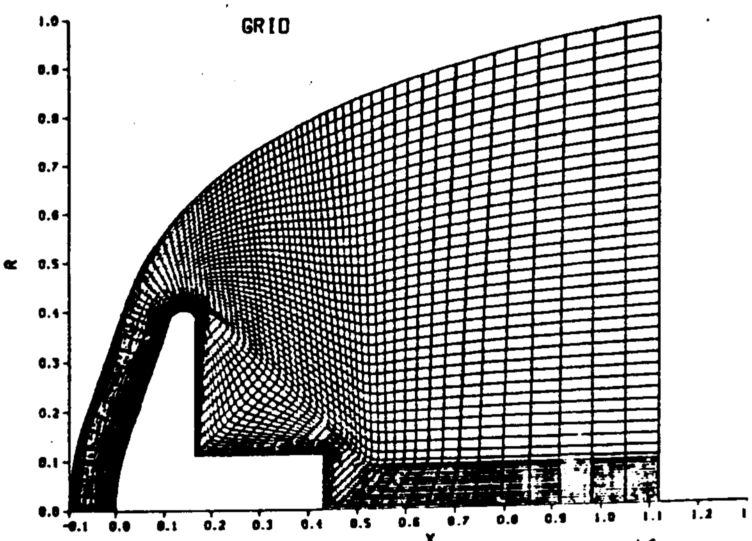
Another problem to avoid is after-body flow impingement, a narrow region around and extending behind the aerobrake where convective heat-transfer becomes very large. The base turning angle [9] is the angle between the free-stream flow vector and the line connecting the frustrum edge with the reattachment. This angle is visible in Figure 2-18. Shih has shown that this angle is about 15° [11]. The best protection against this heating is to keep the structure of the OTV and payload within the "cone of protection" provided by the aerobrake, as measured by the base turning angle.

(a)



Schlieren photo of Mach 13 flight of AOTV model in ballistic range at NASA Ames Research Center (courtesy of Intrieri).

AOTV BLUFF BODY FLOW



(b)

(c)

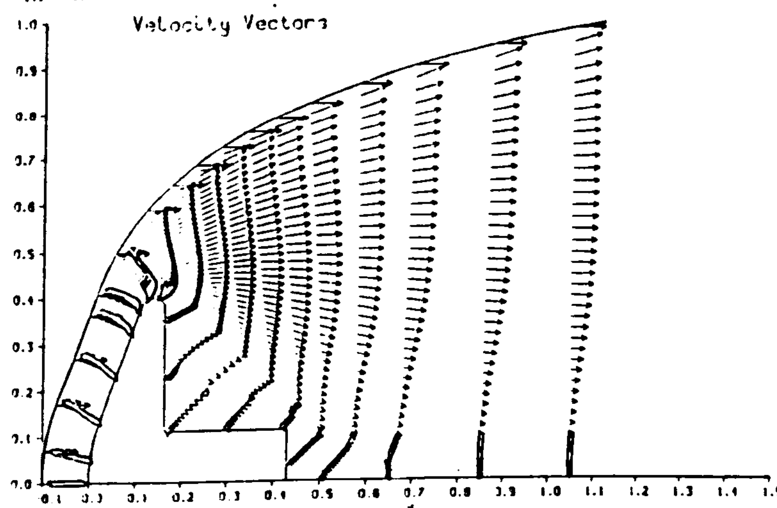


Figure 2-18: Base turning angle of 15° is shown for the after-body impingement.

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Guidance and Navigation [1]

In all past space missions requiring reentry into Earth's atmosphere, only the destination coordinates, expressible with three parameters, have been specified. For the AOTV these three parameters need to be specified; plus the velocity vector, expressible also with three parameters, and the time when the velocity should be attained, are specified at the end of the atmospheric flight. The additional requirement makes the guidance and navigation problem very difficult to solve, and thus renders it one of the most critical of all technological issues.

One must assume that there may be errors in the time, position, and the velocity vector of the vehicle at the time of atmospheric entry caused by unforeseen events. The functional relationship between the position and the velocity vector of the vehicle at the completion and those at the beginning of the atmospheric flight indicates that such an error tends to be amplified: the exit parameters are a sensitive function of the entry parameters. (The beginning and the end of an atmospheric flight can be defined arbitrarily. Typically, the altitude of 150 km is considered to be the border between the atmosphere and the vacuum of space.) Therefore, any such errors must be corrected early during the atmospheric flight. Moreover, the density of the atmosphere at high altitudes, as determined from the Shuttle's flight data, tends to deviate considerably; that is, typically by +/- 25% from the standard values. In order to reach the specified position with the specified velocity despite the fluctuations in the atmospheric density, the vehicle must have a capability for controlling the flight path.

The raked sphere-cone design provides two degrees-of-freedom control by varying lift. There are two methods of controlling lift. In the first method, the angle of attack is fixed, and the direction of the lift vector with respect to the direction of vehicle's motion is changed by varying the bank angle of the vehicle through the use of the Reaction Control System (RCS) engines. By oscillating between two bank angles, the vehicle can achieve a time-averaged L/D which is smaller than the L/D of the vehicle. This method of control is similar to that used in all the pre-Shuttle space missions. In the second method, the angle of

attack is varied as well as the bank angle. The angle of attack can be varied by shifting the c.g.(ie. moving LO_2).

A lifting vehicle induces a coupling between the directional and the lateral motions. Therefore, a roll motion requires use of directional RCS engines as well as lateral RCS engines. This problem has been solved by Gamble in Reference 3.

Because of the complexity of the navigation constraints, it is impossible to define uniquely the most optimum algorithm for guidance and navigation of a lifting AOTV during its atmospheric flight. However, general guidelines can be given:

1. It is advantageous to fly with a negative L/D (with the lift toward the Earth) because this increases the perigee height in exchange for a lengthened duration of the atmospheric flight and thereby lowers the peak dynamic pressure and heat-transfer rates.

2. The cross-range travel (orbital plane change) should be made mostly during the descent phase; the ascent phase should be reserved for correcting for the errors caused by the fluctuation of the air density.

3. During ascent the vehicle should fly near maximum L/D so that if the atmospheric density is too large, the vehicle could roll 180° to produce a positive L/D which will raise the flight path and shorten the flight duration and avoid catastrophic loss of velocity.

4. When the navigational errors and fluctuations in density are such that the vehicle cannot reach the destination orbit, effort should be made to insert the vehicle into the correct orbital plane, sacrificing accuracy in apogee height and phase angle (longitudinal). The vehicle should then execute in- plane rendezvous maneuvers propulsively to correct for the errors.

The worst situation for fluctuation in density is a lower than expected density on descent and then a higher density than expected on ascent. This guides the vehicle into a deeper dive in order to decelerate enough. And when the vehicle ascends it will encounter a very large density, resulting in excessive deceleration. However, calculations show that an L/D of 0.15 would be large enough to lift the vehicle out of the atmosphere on a worst case situation of density fluctuation of +/- 25% [9].

An idea by Menees to aid in navigation is to launch a small projectile in front of the aerobrake shortly before atmospheric entry [7]. By analyzing its trajectory, the density of the atmosphere can be deduced. The density data is then fed into the flight computer as an input to produce a more accurate trajectory prediction and maneuver strategy.

Chapter 3

Engine Specifications

Over the past two decades major research has been conducted in an effort to produce a rocket propulsion system capable of reliable and efficient transportation of payloads into and from Earth orbit. Until recently, the mainstay of NASA was the RL10A-3-8. This engine was defined in 1967 as the engine for an imporved Centaur. The RL10A-3-8 was used in the early Shuttle Upper Stage Studies, and it was found to be lacking in several areas [3]. In an attempt to produce a state-of-the-art high performance engine, study contracts were awarded to Aerojet, Pratt & Whitney, and Rocketdyne to determine what could be done to improve upon current designs. This action instigated independent research into the development of modern light-weight high performance engines.

The engine type which came out of this research was the Category IV expander cycle engine. This engine was the first expander cycle engine specifically designed for the OTV mission requirements. Many of the features designed for this engine have been carried through multiple design iterations to the present Pratt & Whitney advanced engines. At the time of its design, the Category IV engine maintained the highest chamber pressure (915 psia) thought possible for existing materials [3].

The advent of modern turbomachinery design in the 1980s has permitted the stresses acceptable to modern engine chamber designs to be nearly twice that of earlier engines. As a result of this advance in technology, NASA has re-evaluated the requirements it is placing on the technology goals of the OTV engine. To this date, no engine design has met all of the requirements set out by NASA. The Pratt & Whitney 1985 Advanced Expander Cycle Engine, specified the RL100, shows the most promse in fulfilling the mission requirements currently set down for a manned OTV mission. Table 3-1 shows the 1987 updated goals for the OTV engine in comparison with the specifications of an unmodified stock RL100 engine [1].

Table 3-1

Comparison of 1987 OTV Engine Goals and the RL100 Engine

Parameters	'87 NASA Goals	RL100
Man-rating	Yes	Yes
Fuel	Hydrogen	Hydrogen
Oxidizer	Oxygen	Oxygen
Vacuum Thrust	7500 lbf (per)	7500 lbf (per)
Engines per Vehicle	2 Minimum	2
Mixture Ration O/F	6.0	6.0
Mixture Ratio Range	5-7	5.5-6.5
Inlet Temperature-		
Hydrogen	37.8 °R	(TBD)
Oxygen	162.7 °R	(TBD)
Aerobraking Design Criteria	The engine must be compatible with aeroassist return of the vehicle to low-Earth orbit.	
Vacuum Specific Impulse	490 lbf-sec/lbm	477 lbf-sec/lbm
NPSH-		
Hydrogen	15 ft-lbf/lbm	15 ft-lbf/lbm
Oxygen	2 ft-lbf/lbm	2 ft-lbf/lbm
Weight	360 lbm	290 lbm
Length	(TBD)	60 in.
Reliability	.9997	(TBD)
Operational Life	20 hours	(TBD)
Service Free Life	4 hours	25 missions

An unmodified RL100 meets or exceeds most of the requirements stipulated by NASA for the technology goals of the OTV engine. The chamber pressure (1210 psia) and the vacuum specific impulse of the RL100 are limited by the reduction in efficiency inherent in using small pumps [1]. Research is currently being conducted in an effort to alleviate the limitations of the smaller pumps by improving the purity of the materials used in the production of the pump shaft, seals, bearings, gears and thrust chamber. Advancements and innovations in this area can be expected to raise the overall performance of the stock RL100 by a minimum of at least five percent. In an attempt to compensate for the performance limitations experienced by the RL100, several design innovations have been incorporated.

An extendable nozzle is incorporated into the engine design to allow a large expansion area ratio without the corresponding length requirements for storage and transportation. The extendable nozzle of the RL100 produces an increase in specific impulse of approximately 20 lbf-sec/lbm over the same engine equipped with a stationary nozzle [3]. Figure 3-1 shows how the extendable nozzle of a RL10 derivative engine functions to increase the expansion area ratio without increasing the overall length of the engine.

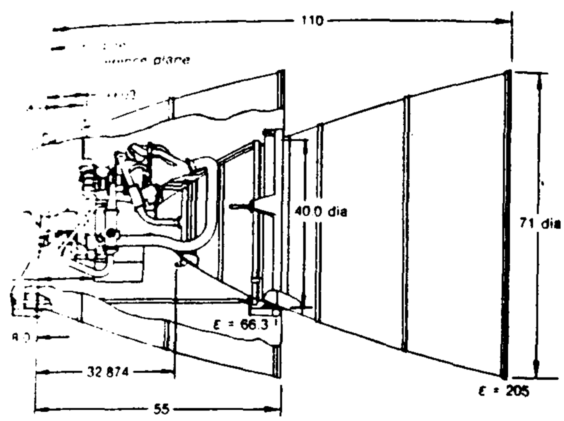

Figure 3-1: RL10 Derivative Engine

The extendable nozzles produced by Pratt & Whitney are composed of carbon/carbon fibers coated with silicon carbide. The use of these modern thermal resistant materials over traditional nozzle materials increases the operational life of the engine while also decreasing its weight. By being a radiation-cooled nozzle rather than a dump-cooled nozzle, the complexity and size of the engine and pumping system is reduced.

The turbomachinery of the RL100 will be manufactured using state-of-the-art technology to permit the pumps to perform at a maximum output of 150,000 rpm [3]. Figure 3-2 shows the flow schematic of the RL100 engine at full thrust. The performance requirements of the gears and turbines are shown here to be well above any engine with similar performance ratings. By using high speed pumps, the overall mass and displacement of the RL100 is reduced by one-third when compared with similar engines.

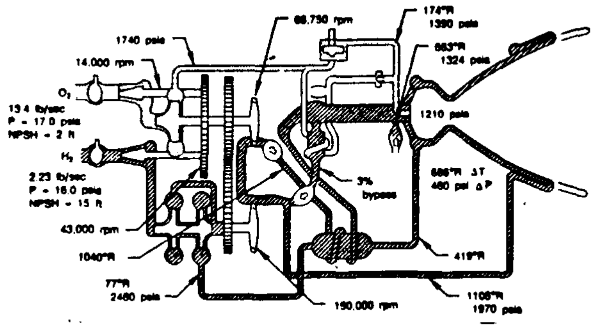

Figure 3-2: RL100 Engine Flow Schematic

Unlike comparable engines, the RL100 is self-contained and modular. This allows the engine to be easily separated from the OTV for inspection and maintenance [5]. As seen in Figure 3-2, the simplicity of the fuel transfer system requires that only two valves be shut to isolate the engine from the fuel delivery system [3]. While in free- fall, the engine can be removed from its support structure in the aerobrake and disconnected from the avionics of the OTV in approximately 3 hours [5].

For the reasons stated above and the stipulated mission requirements, it was determined that two RL100 engines with a combined thrust of 15,000 lbf would be the best main propulsion for a man-rated OTV mission. Two engines were chosen to give the OTV single-engine-out propulsion capability. Current research conducted by Pratt & Whitney and Aerojet has shown that a reliability of 99.6% can be expected on a vehicle with two engines. This data gives a nonindependent failure rate between 0.03 and 0.05 [6]. The fuel efficiency obtained by using two engines is less than that obtained using a single engine, but the reliability and safety gained from a two engine design increases the expected life of the OTV.

During the aeroassisted deceleration, the lift versus drag characteristic of the aerobrake will be changed by rotating the oblate aerobrake about its center of gravity. Having the engine nozzles extended would generate problems with the aerodymanics and cause severe deterioration of the nozzles themselves. Figure 3-3

shows how the engines are retractable flush to the aerobrake. The engines are inherently capable of tolerating the temperatures at the stagnation point in front of the aerobrake without oblation. This additional factor makes the RL100 engine the ideal main propulsion system for WWSR's man-rated OTV.

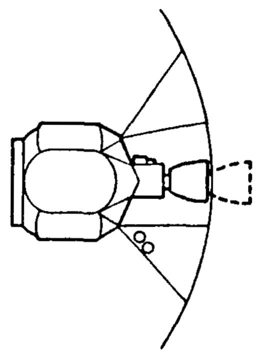

Figure 3-3: Aerobrake Engine Configuration

Chapter 4

Fuel System

The fuel system for the OTV will consist of six pairs of fuel/oxidizer tanks each with independent delivery and pressurization systems. Check valves will be incorporated into the delivery systems to allow isolation of each tank and permit pressure relief when necessary. Figure 4-1 is a schematic of the fuel system showing the check valves, delivery systems and pressurization systems for each tank and the entire system.

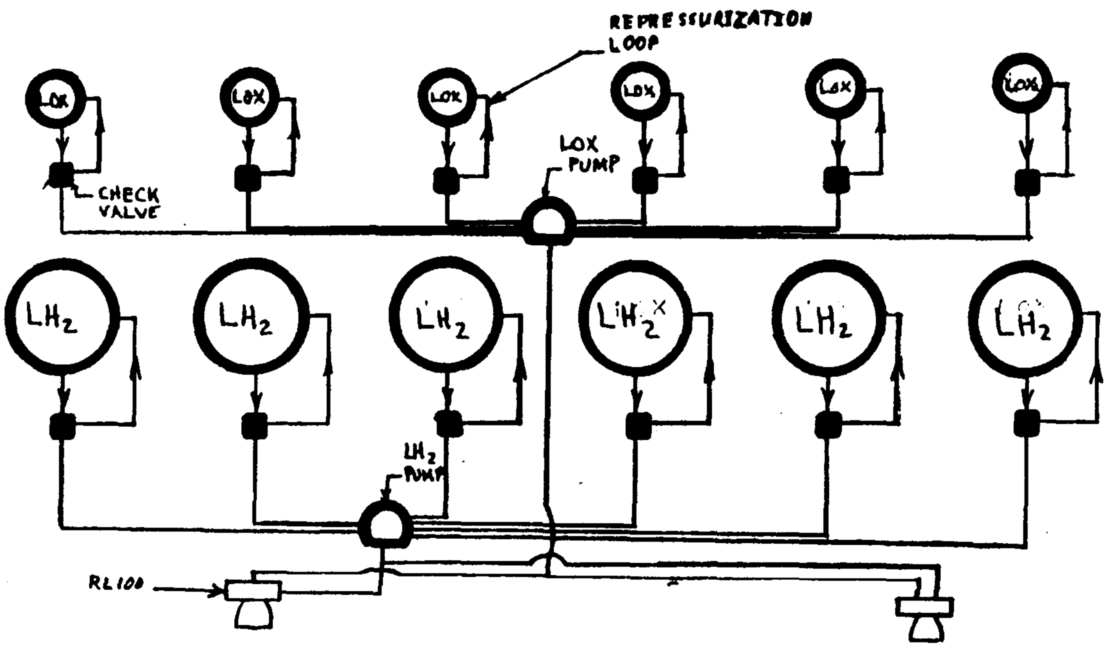

Figure 4-1: Fuel System Schematic

The fuel/oxidizer delivery system will independently draw and pump from each tank. The fuel/oxidizer will then be pressurized at a second pumping station just prior to entering the engine. This allows the fuel/oxidizer to be brought from its containment pressure of 7 psia to the inlet pressure of 17 psia for oxygen and 16 psia for hydrogen. The internal pumps of the RL100 then increase the pressure of the reactants to over 1200 psia before they reach the combustion chamber.

It will be necessary to have the fuel tanks pressurized at a constant level to

simplify the pump and turbine requirements of the delivery and propulsion system. The reactants that boil off, and are usually vented, will be used to maintain a constant pressure with the tanks. This will be accomplished by computer controlled venting and recycling of the gaseous reactants. Any excess oxygen will be shunted to the ECLSS to be used inside the manned module. Any excess hydrogen will have to vented to space. Since it is unsafe to mix the reactants during storage, separate pressurization systems will have to exist for each tank. Figure 4-2 is a schematic of the pressurization system for one pair of tanks with it connection to ECLSS.

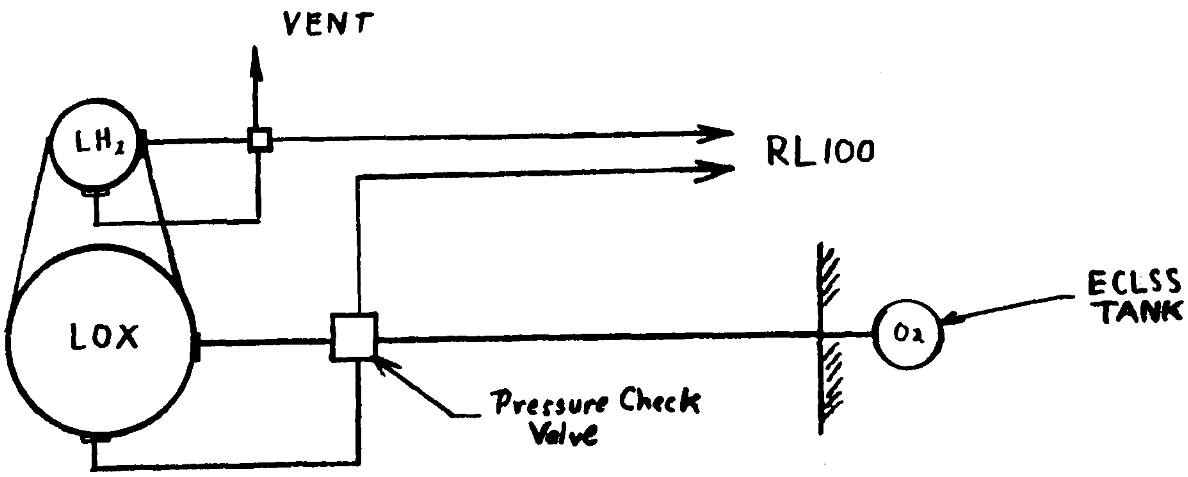

Figure 4-2: Pressurization System of Tank Pair

The reactants used in the OTV's fuel system will be liquid oxygen and liquid hydrogen. Typical specific impulse values for the RL100 using these reactants would be between 470 and 485 seconds [6]. The reactants efficiency will be improved by addition of metallic aluminum suspended within the liquid hydrogen and the addition of a extendable nozzle to the main engine structure. The efficiency obtained after these modifications will increase the specific impulse of the RL100 to approximately 502 seconds [1] not assuming any design improvements during its production.

In determining the fuel requirements to fulfill the mission objective, a dry mass for the OTV of 32,670 lbm was used. The requirements of the mission

(worst case scenario) are to transport a 24,000 lbm payload form LEO to GEO and then return to the Space Station without a payload. In this case the payload is considered to be part or all of an orbiting space platform. For this mission, the engines are required to fire multiple times, assuming instanteous acceleration, to facilitate orbital transfer, course corrections and aeroassisted deceleration. This information is covered more in depth in Appendix 2.

The fuel requirements for the mission were determined by using the rocket equation and iterating backwards through the required velocity changes. A total fuel mass of 121,616 lbm is required to produced the necessary transfers for this mission. For safety the tanks will be filled to 125,000 lbm. However, if necessary, the tanks could be filled to capacity, 132,000 lbm. The fuel is separated in 18,857 lbm of liquid hydrogen and 113,143 lbm of liquid oxygen.

Chapter 5

Materials and Structures

The basic ideas in designing the material and structure for the OTV is to make it lightweight, strong enough to withstand the stress of aerobraking, and heat and radiation resistant.

Lightweight - If the structure is reduced by one pound, then the savings in fuel will be four pounds. Therefore, the structure will be made as light as possible in order to save fuel or to increase the payload.

Strength - Since the OTV is space based, the structure will not have to be as heavy (strong) as a ground based structure. The acceleration experienced during the ascent to GEO is at least an order of magnitude lower than the 1.5 g deceleration of aerobraking [6]. The structure will be under maximum stress during aerobraking, not the ascent to GEO. The stress caused by thrusting the engines is also in the same direction as the stress of aerobraking because the engines protrude from the heat shield. The structure is designed to withstand the stress of aerobraking (1.5 g) and a safety factor of 1.4, unless stated otherwise.

Heat resistance - The structure will be exposed to thermal cycling and high temperatures. During aerobraking the structure will be exposed to various high temperatures depending on the location of the part. All exposed areas must retain sufficient strength at the maximum temperature to withstand the stress that occurs during the exposure at that temperature. Because the OTV will be used many times, the structure must be able to cycle between maximum and minimum temperature without losing a critical amount of strength. Thermal cycling will also occur due to exposure to solar radiation on one side, while the other side is shaded. Temperatures could cycle from -175° to +500°F if the OTV is not constantly rotating or if the solar radiation is not reflected.

Radiation resistance - Solar radiation tends to weaken materials. The

structure must be designed to retain the required strength during the OTV's entire lifetime, or the parts must be easy to replace.

Structure - The structure includes the engines, engine quick disconnect plate, thrust structure, connectors, tanks, tank support rings, struts and supports, command module, EVA module, docking/service/equipment/avionics assembly, payload attachments, robot arm and aerobrake. A description of each follows.

Engines - The engines have extendable nozzles that protrude out the heat shield. During re-entry the nozzles will be retracted so that they are flush with the heat shield. Refer to Chapter 3 for a more detailed description of the engines. Mass = 580 lb.

Engine quick disconnect plate - This aluminum plate enables the engines to be disconnected quickly for repair or replacement. Mass = 100 lb.

Thrust structure - The thrust structure transmits loads from the engine to the rest of the structure and to the payload. The assembly consists of a cone-frustrum-shaped composite structure consisting of honeycomb sandwich skin panels (0.01 inch graphite/epoxy face sheets on a 0.079 inch thick nomex core of 0.91 lbm/ft^3 density), a thrust distribution ring, and thrust beams. The assembly begins directly below the command module and attaches to the tanks through the connectors. Six tubular thrust beams (2 inches in diameter) are attached to the aerobrake to uniformly distribute the load across the brake. Total mass = 210 lb.

Propellant tanks - A spherical design has been chosen because it is simple, has good pressurization characteristics, and has maximum volume-to-mass ratio. The tanks can be spin formed and then chem milled to the correct thickness [7]. The tanks will be insulated by multi-layered insulation (MLI) which is described in the next section. Unlike ground-based vehicles, a space-based OTV is designed to operate solely in the vacuum of space and does not require that propellant tank pressures be maintained above 14.7 psia. The propellant will be held at a low pressure, 7 psia, to reduce the load on the tank structure, therefore making the structure lighter [7]. Figure 5-1 shows that the weight of the propellant tanks

decreases as the tank pressure decreases. Reducing tank pressures below atmospheric requires that propellant saturation conditions be lowered so that the fluids remain in the liquid phase [7]. The LO_2 tank is not pressure cycled (purged) between missions, but the LH_2 tank must be. The tank interiors are designed to support slosh baffles, inner bladder and a liquid acquisition device.

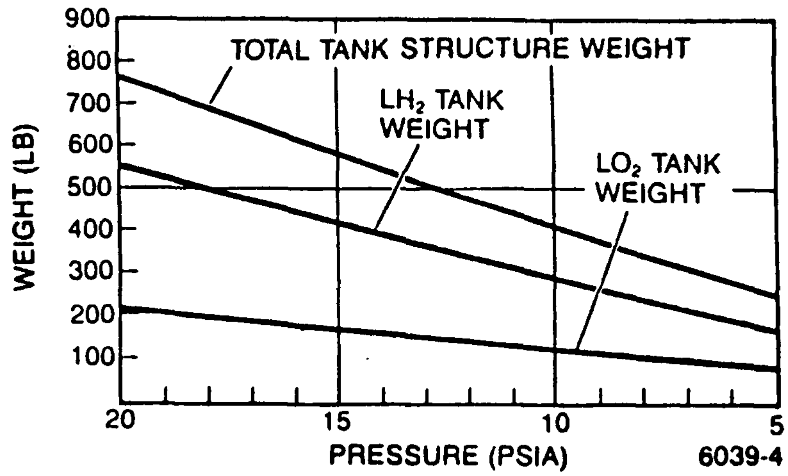

Figure 5-1: The relation between tank pressure and structural weight.

The selection of material for the propellant tanks is important because many materials are sensitive to LO_2 and many can be embrittled by hydrogen. Also, low temperatures can reduce ductility and fracture toughness of some metals. 2219 aluminum is often used for cryogenic tanks and works well. A new alloy, 2090-T8E41 aluminum alloy (Al-Cu-Li-Zr), has been developed that has better properties than 2219 [7]. Table 5-1 is a listing of mechanical properties of the 2219 and 2090 alloys. The 2090 alloy has higher strength and less density. The higher strength-to-weight ratio will enable the tanks to be lighter. Also, the 2090 alloy's tensile strength increases at lower cryogenic temperatures [5]. Table 5-2 lists the mechanical properties of 2090 at 298, 77, and 4°K showing this increase in tensile strength.

Table 5-1

Properties of 2219 and 2090 aluminum alloys.

Property	E	Ftu[1]	Density	Fty[1]			K_{IC}		
Units	MSI	KSI	$lb/in.^3$	KSI			KSI × $(in.)^{1/2}$		
Temp (F)	70	70	70	70	−320	−423	70	−320	−423
2090-T8E41[3]	11.0	82	0.092	77.6	87.0	89.2[2]	31	47	59[2]
2219-T87	10.5	63	0.102	56.0	66.9	74.3[2]	33	39	44[2]
2219-T62	10.5	54	0.102	36.0	42.8	48.6	32	35.5[4]	35.5[4]

Table 5-2

Fracture Toughness and Tensile Properties of 2090-T8E41 at 298 K and 77 K

Temperature	Fracture Toughness L-T	L+45	S-T	S-L	Uniaxial Tensile Properties Yield Stress (MPa)	UTS (MPa)	% Elong. (on 25.4 mm)
	(MPa$\sqrt{m}$)						
298 K	36 35#	29	16	17	535	565	11
77 K	51+ 51#	47+	13	15	600	715	14
4 K	64#	-	-	-	615	820	18

The tanks were designed so that six pairs will carry the fuel necessary for maximum payload (worst case scenario). Each LO_2 tank is 4.2 ft radius, 100 lb, and holds 18856 lb of LO_2. Each LH_2 tank is 5.8 ft radius, 250 lb, and holds 3144 lb of LH_2.

MLI - The Kapton MLI is composed of layers of 3.75 micron aluminized kapton plastic (30 for LH_2, 20 for LO_2) each separated by a silk-net layer. See Figure 5-2. These are held together by widely spaced plastic pins. The tank is built like a Thermos flask, with an evacuated double wall. There is no efficient way for heat to be exchanged between the layers. If the heat follows a tortuous

path by way of the plastic pins, the low heat conductivity of the plastic allows very little to get through. And if it radiates from layer to layer, the aluminum coating on each sheet reflects nearly all of it. This insulation allows liquid hydrogen and oxygen to be store for extended periods of time in space. MLI has been tested in a vacuum at NASA Lewis [1]. MLI was chosen over the commonly used polyurethane foam insulation to reduce volume and weight. MLI also offers some protection from meteoroids/debris.

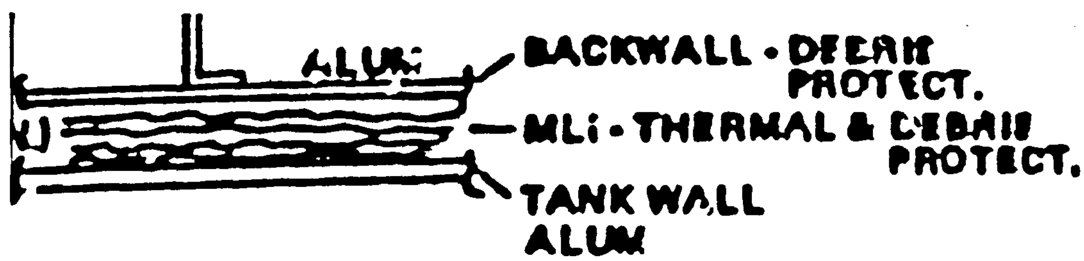

Figure 5-2: Schematic of MLI Thermal Protection

Connectors - The connectors attach the LH_2 tanks to the thrust structure and the LO_2 tanks to the command module. These connectors contain the disconnect panels that allow the tanks to be modular. Up to six pairs of tanks can be added to the OTV. This is shown in Figure 5-3 and 5-4. An aluminum alloy is used because the tanks will be connected and disconnected often. This handling might damage a composite material and cause delamination. For each pair of tanks (LH_2 and LO_2) two polygonal frames of aluminum support all the propellant system plumbing and interface with the propulsion system. Male connectors are located below the LH_2 tank and above the LO_2 tank. Female connectors are located at the thrust structure below the command module and at the top of the command module. Mass = 75 lb per set.

Tank support rings, struts, and supports - These components will be RCA-2606114 graphite/epoxy. This is an ultra-high modulus graphite unidirectional tape/low microcracking epoxy. The material's resin solid content is 40% with a nominal prepreg thickness of 0.0025 inch [3]. The low thermal conductivity of this

material will prevent heat loss from the cryogenic tanks. The ceramic graphite/epoxy also has a much greater strength-to-weight ratio than metals, allowing the structure to be lighter. Figure 5-5 shows the ultimate tensile strength of the G/E.

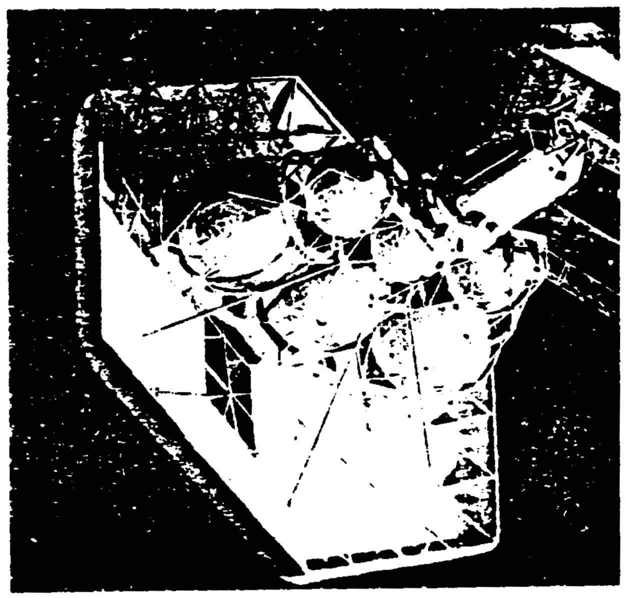

Figure 5-3: Drawing of OTV with two modular tank sets connected to the central structure.

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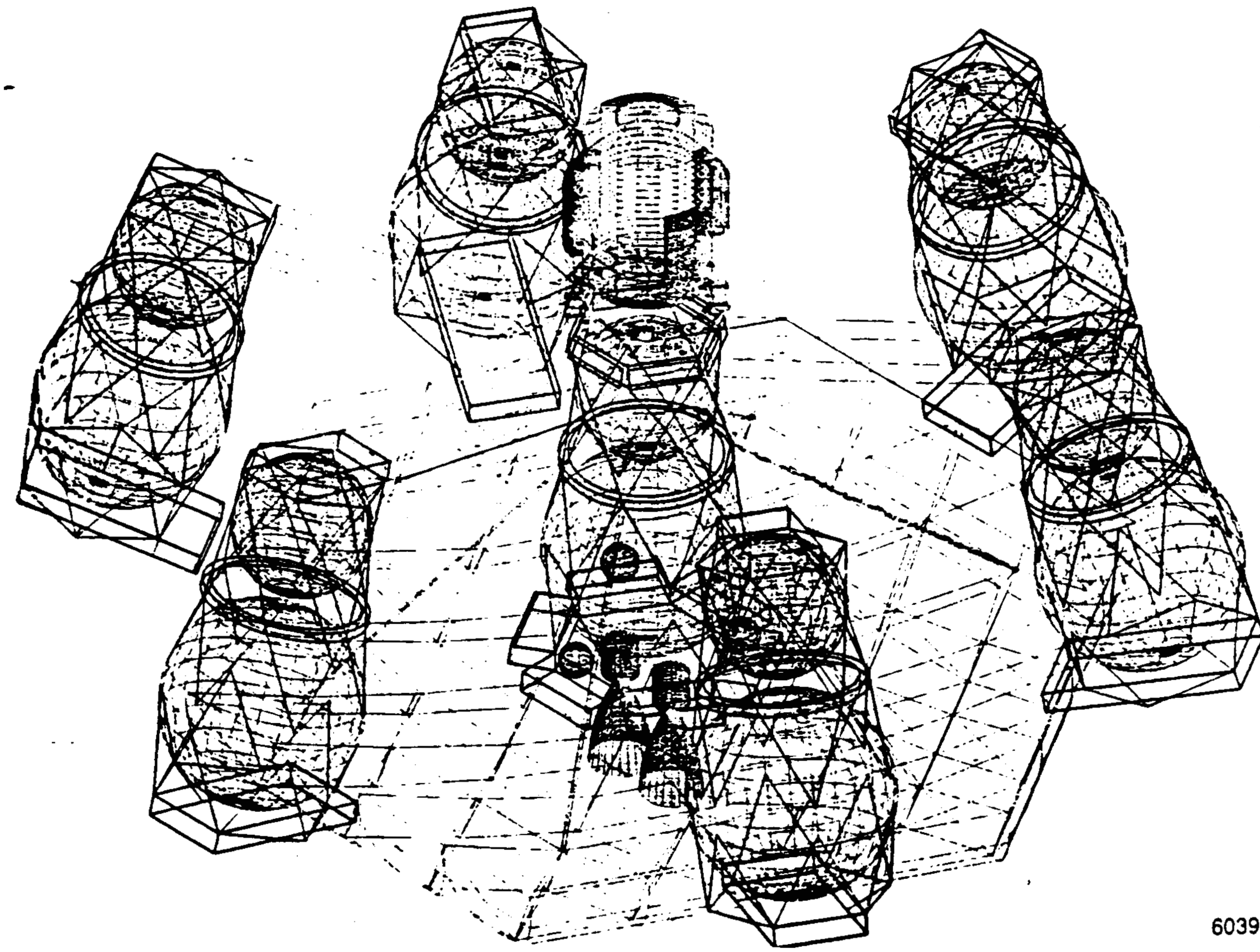

Figure 5-4: Tank support structure and connections with the central structure.

RCA has performed thermal cycling tests, radiation tests and combinations of the two tests resembling 10 years at GEO orbit, and has found that this graphite/epoxy retains much of its strength. The radiation environment was simulated by exposing the test coupons of the materials to an electron beam of energy 12 MeV to a total ionization dose of 3×10^8 rads. The dose rate during irradiation was 3×10^8 rads/hour. This dose rate is about four orders of magnitude higher than the space dose rate and represents the worst-case simulation of the space radiation environment [3]. The thermal environment was simulated by thermal cycling (3000 cycles) between temperature extremes of -300 and 160 °F. A transition rate of about 11 °F per minute was used for thermal cycling [3]. The results showed that beginning of life tensile strength = 135 ksi, end of life tensile strength = 110 ksi [3]. The tensile strength of aluminum is closer to 50 or 60 ksi.

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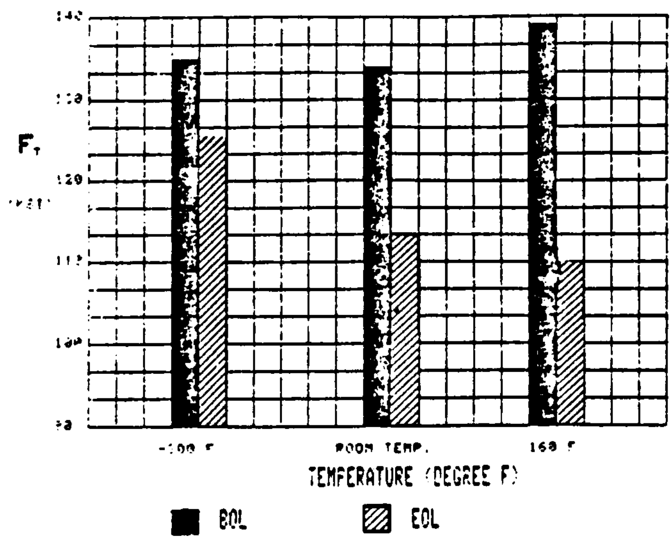

Beginning-of-Life and End-of-Life Longitudinal Tensile Strength of UHM Graphite/Epoxy Composite

Figure 5-5

Graphite/epoxy tubular struts (2 inch diameter) are used to attach the tanks to the tank frames. Figures 5-4 and 5-6 show the skeleton structure of the OTV. Twenty-four struts are used for each tank (see Figure 5-6). To prevent buckling of the tank wall, strut angles must be selected such that the tank does not experience negative deformations or compressive stresses. A G/E tubular support ring (5 inch diameter) will support and separate the two tanks. The mass of these components is 220 lb (for a pair of tanks).

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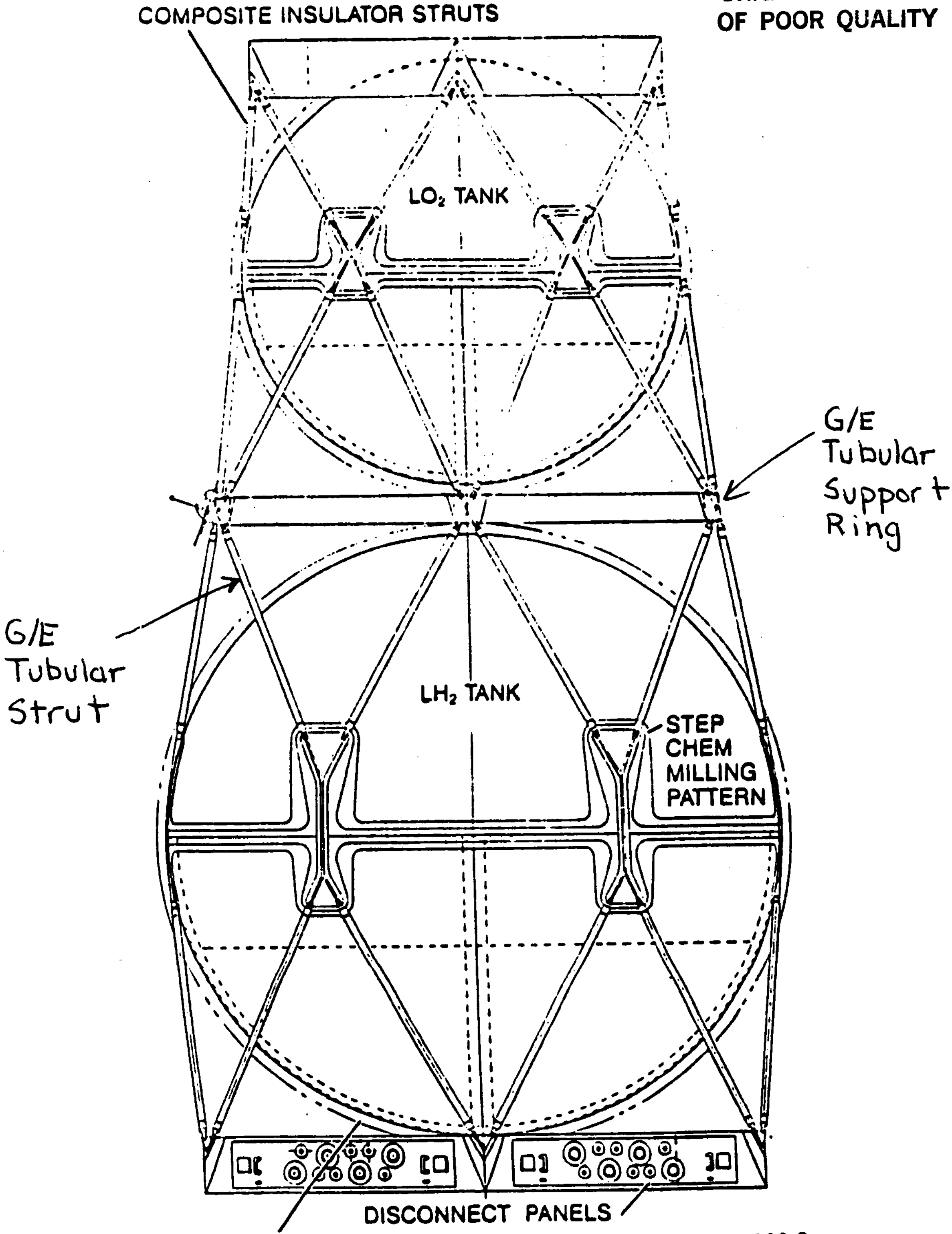

Figure 5-6: Tank Support Structure

<u>Command/EVA module</u> - The modules will be semimonocoque. The outer skin will be stiffened with ring frames and skin stringers (see Figure 5-7). The structure will be 2090 aluminum. The command module also holds the graphite/epoxy tube ring support and the female connector for the propellant tanks. The mass for the structure of the command and EVA modules are 10700 and 1500 lb, respectively. The modules also contain three hatches with a combined mass of 300 lb.

<u>Docking/service/equipment/avionics assembly</u> - This assembly will be attached to the side of the EVA module. The assembly provides for external mounting of equipment and avionics, a universal docking system, and service connector panels for fluids, gases, and electric power. A peripheral latch/release system for payload accommodation and robot arm are attached to the top of the EVA module. The arm is discussed in Chapter 12. Mass = 180 lb (excluding the robot arm).

<u>Aerobrake</u> - The aerobrake will have to repeatedly withstand very high temperatures and heating rates for a short period of time, and keep the temperature of the structure below 350°. For a one pass mission the maximum temperature on the surface may reach 2000° and maximum heating rate could reach 35 to 40 W/cm^2. Therefore, the OTV will conduct multiple pass missions, thus reducing the maximum surface temperature to below 1000° and maximum heating rate to 20 W/cm^2. The aerobrake is discussed in detail in the aerobraking chapter. Mass = 2800 lb.

<u>Heat and debris protection</u> - The OTV structure and payload need to be protected from the heat of aerobraking and collisions with meteoroids/debris. A very thin aluminum foil extendable blanket will be used to surround the structure and payload. The high reflectivity of the aluminum foil will reflect most incident solar or heat flux away from the OTV, and will provide some protection from space debris. The probability of puncture by micrometeoroids is low and could be substantially reduced if the OTV were to be stationed within a depot when not in use [2]. And even if a micrometeoroid did puncture a fuel tank, the tank would leak but would not fail catastrophically [4]. Therefore, a heavy meteoroid protection shield will not be used.

Surface coating - All surfaces that will be exposed to solar radiation and radiative heat transfer from the aerobrake maneuver will be painted white (excluding the heat shield tiles). White paint (293 and S13GLO) has the best reflectivity and lowest absorption. The absorptance is 0.18 and the emittance is 0.9. This reduces the amount of solar radiation that is changed into heat.

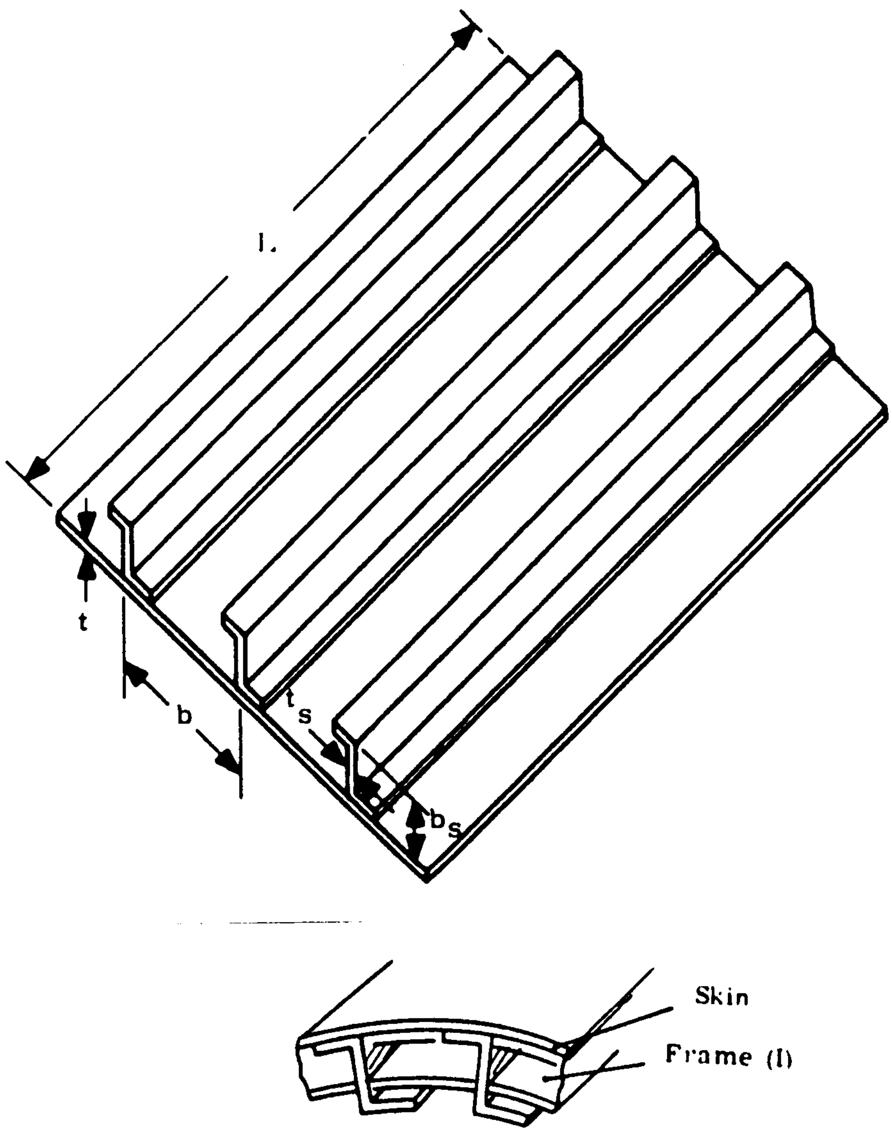

Figure 5-7: Semi-monocoque Structure

$t = 1.0$ in
$t_s = 0.5$ in
$b = 10.0$ in
$b_s = 1.5$ in

Chapter 6

Ambient Heat Transfer

The ambient radiation received by an object at one astronomical unit from Sol is known as the solar constant and has a value of 1353 W/m^2. this value assumes the object to be located at the equator of the Earth and perpendicular to the incident radiation. The actual value received by the OTV will be within 15% of the solar constant [2]. The OTV will also radiate excess heat to the surroundings. Conductiion and convection can not occur into a vacuum, therefore, radiation will be the only way for the vehicle to lose heat.

The energy flux lost to the surroundings by radiation can be determined by using the equation:

$$q'' = e \times o \, (T^4{}_s - T^4{}_{surf}) \quad [2]$$

In this instance the ambient temperature of the surroundings is approximately 4 °K not including the Earth. The surface temperature of the OTV is limited by temperatures of the fuel and manned module. Assuming conduction from the engine and crew quarters through the support structure of the vehicle, the surface temperature of the vehicle would be at a maximum of 295 °K for GEO conditions. This value can reach as high as 350 °k during aerobraking [3]. The average emissivity value for the OTV materials is 0.89 [2]. Assuming this value, the radiation flux to the surroundings is 80.740 W/m^2. The OTV is receiving 16.75 times as much energy as it is radiating.

This influx of energy will cause a loss in fuel due to boil-off. To partially alleviate this problem, the OTV will be coated with materials that overall have a low transmissivity and absorptivity while maintaining a high reflectivity. The relationship of these three values can be seen in the following equation:

$$p + a + t = 1 \quad [2]$$

Where p is the reflectivity, a is the absorptivity, and t is the transmissivity.

Polished aluminum, aluminum coatings, or gold will be used to insulate areas (fuel tanks) where radiation absorption is to be kept to a minimum. Those areas (aerobrake and exhaust nozzle) where radiation emission is required will be coated with silicon carbide and ceramic tiles similar to those used by the Space Shuttle. The manned module will be constructed of aluminum with a white metallic coating. Since this module is surrounded by six sets of fuel tanks, this coating will be all that is required to maintain a minimum absorption of energy. The combination of these materials will allow the vehicle to maintain a relatively constant temperature for the crew compartments and the fuel tanks. From data already obtained, the expected surface temperature of the OTV will be approximately 200 °K [1].

Chapter 7

Electrical Power System

The Electrical Power System (EPS) produces electrical power for the OTV during all mission phases. The EPS onboard the WWSR OTV will consist of two hydrogen (H_2) - oxygen (O_2) fuel cells and one bipolar nickel-hydrogen battery. The fuel cells will be United Technologies' latest version of Shuttle-technology power plants (which are thirty percent lighter than current cells). [1] These fuels cells are extremely reliable and provide the most effecient means of production of electricity for the OTV's mission (two week duration at 20 kilowatts maximum). The Ni-H battery represents state-of-the-art technology in energy storage. It is the lightest, most reliable, and most powerful of all spacecraft battery systems.

The fuel cells produce direct current electrical power through a controlled chemical reaction of the hydrogen and oxygen. The hydrogen and oxygen reactants will be cyrogenically stored in the main tank sets. Proper reactant gas pressure is maintained in the tanks by small heaters controlled by the onboard computer system. Additionally, the oxygen tanks will double as the storage tanks for the life support systems. The fuel cells will simultaneously produce 28 volts of direct current at a maximum power of 10 kilowatts. The total maximum, onboard power requirements are 7.5 kilowatts; the extra capacity is available to power the OTV's payloads if needed. The cells will be actively redundant, as each cell is capable of providing full mission power in the event that one goes off line. Power production is controlled by the Electrical Control Unit (ECU) which is part of the fuel cell. The ECU controls the reactant flow rate as determined by the power demand. A by-product of the production of electricity by the reaction of hydrogen and oxygen is pure water. This water, on the order of 6 kg per hour, will be stored in one of two water tanks and can then be used for thermal control or human consumption. Total mission energy is expected to be approximately 2000 kW-hours thus requiring about 1390 lbs kg of oxygen and 210 lbs of hydrogen [1].

A single nickel-hydrogen battery will provide emergency power backup, line

transient supression, and autonomous startup capability. The Ni-H battery will be fully charged at the Space Station and will be actively recharged by the onboard fuel cells during a mission. The battery will be capable of providing reduced emergency power for approximately two hours in the event of a catostophic failure of the fuel cell system (a source of electricity is needed to start or restart power production in the fuel cells). It's main function, however, is to provide a source to smooth power surges caused by major subsystems coming on line [3].

Electrical power distribution is controlled by the Electrical Power Distribution System (EPDS). The EPDS converts and controls the flow of electricty to the subsystems of the OTV. Additionally, the EPDS monitors and controls the reactant gas levels and pressures, surge supression, and charging of the Ni-H battery. The EPDS is connected to the Data Management System for status output and crew control.

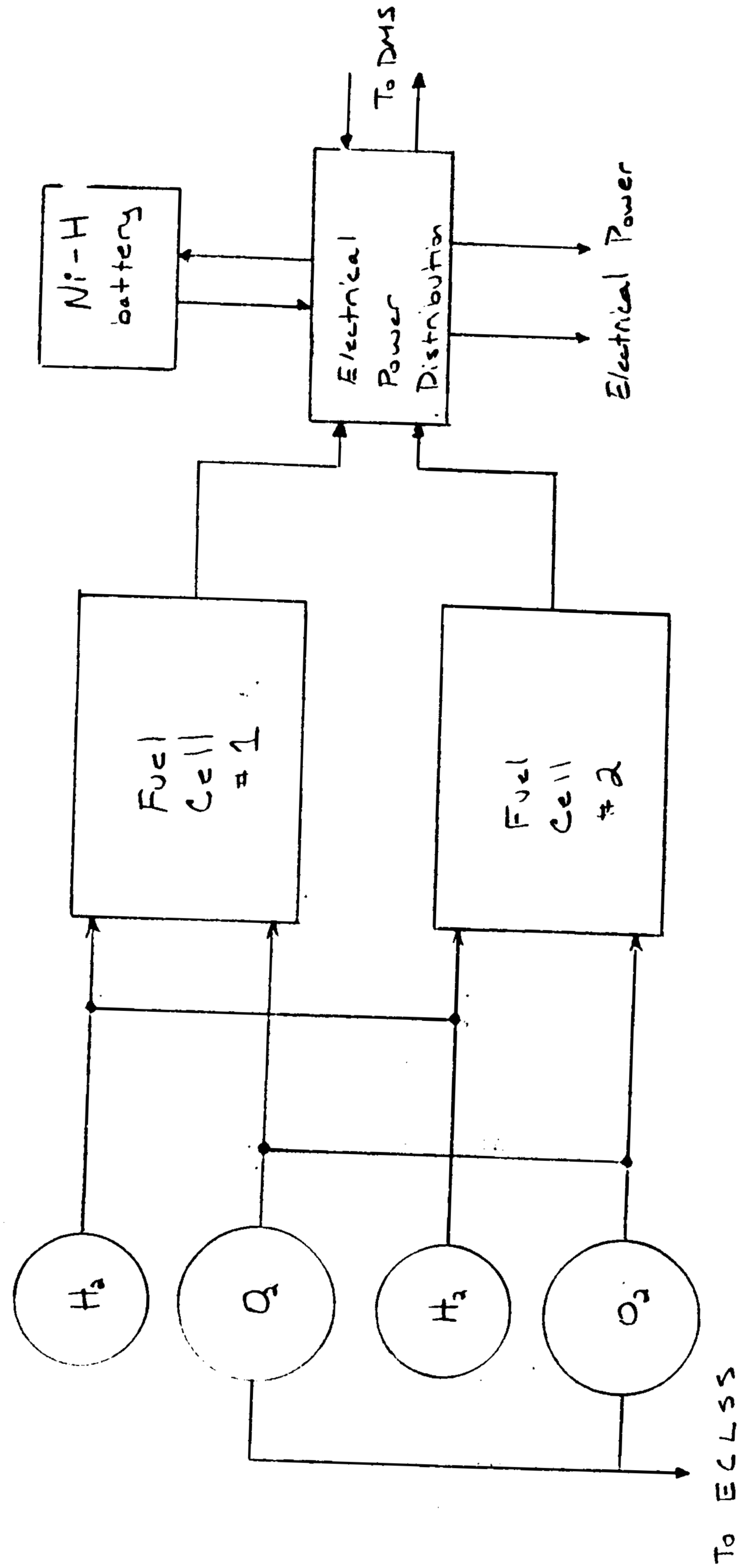

Figure 7-1: The Electrical Power System

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Chapter 8

Environmental Control and Life Support Systems

The ECLSS for the OTV will be largely based on present technology used on the Space Shuttle [1]. This was decided to eliminate the cost of research and development on new systems. Also, present ECLSS technology on board the Space Shuttle has proven to be highly reliable. The life support will have the following systems [4]:

1. Atmospheric Revitalization
2. Thermal Control
3. Crew Systems

A system integration flow chart of the above systems is shown in Figure 8-1. Each of these systems will be discussed in more detail below.

Atmospheric Revitalization

This system is given the task of providing fresh air to the crew members and is therefore the most important system. It is illustrated in Figure 8-2.

Air is drawn into the system by fans located strategically throughout the crew and command modules particularly around the cockpit area. After passing through the intake ducting, the air is filtered by a debris trap to remove dust and foreign particles. The exiting air is then divided into several other air streams which are individually processed. One stream enters a unit of canisters containing lithium hydroxide, copper sulfate, and activated charcoal. The lithium hydroxide extracts carbon dioxide and the charcoal removes air impurities for odor control. The copper sulfate extracts ammonia.

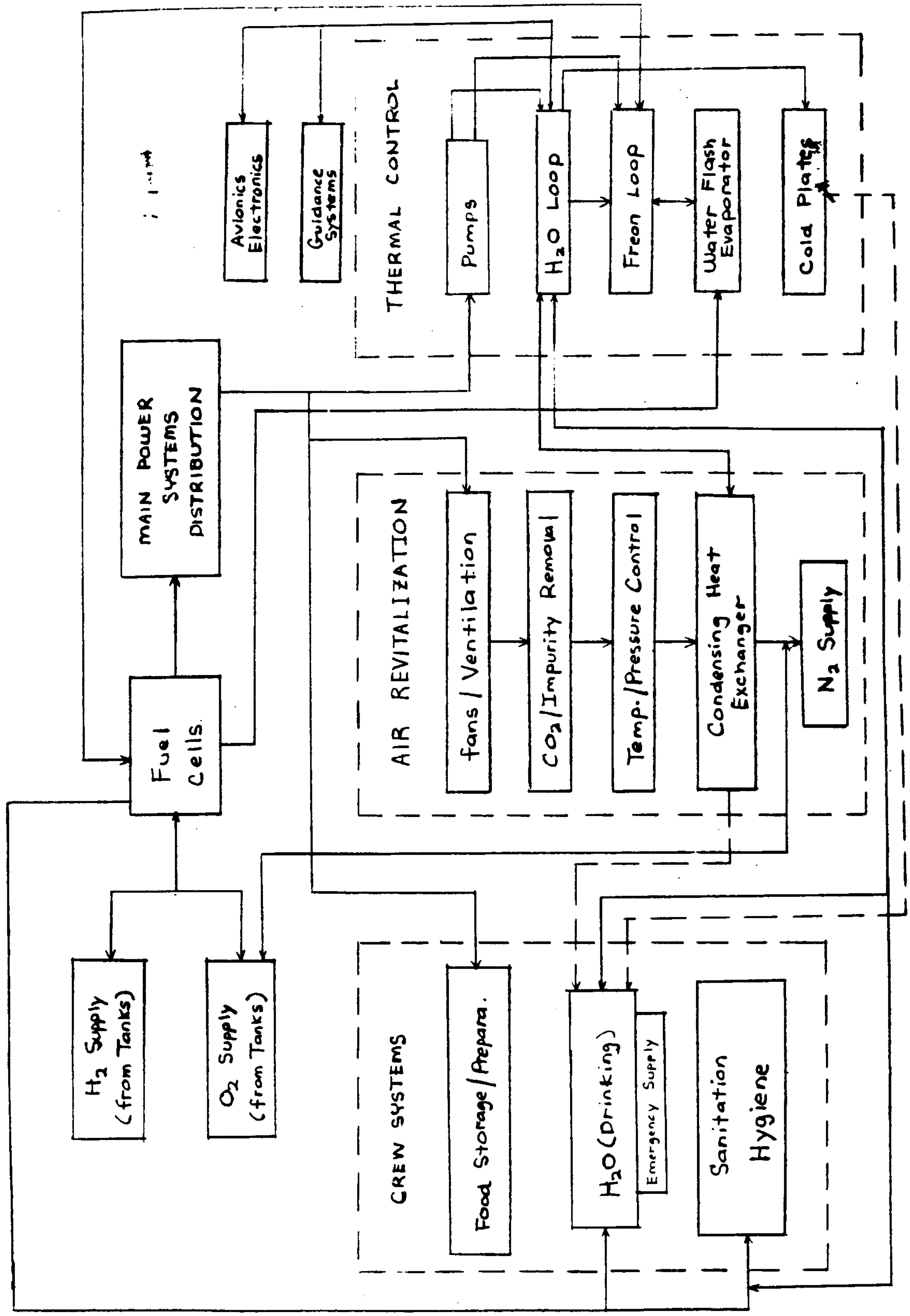

Figure 8-1: OTV ECLSS

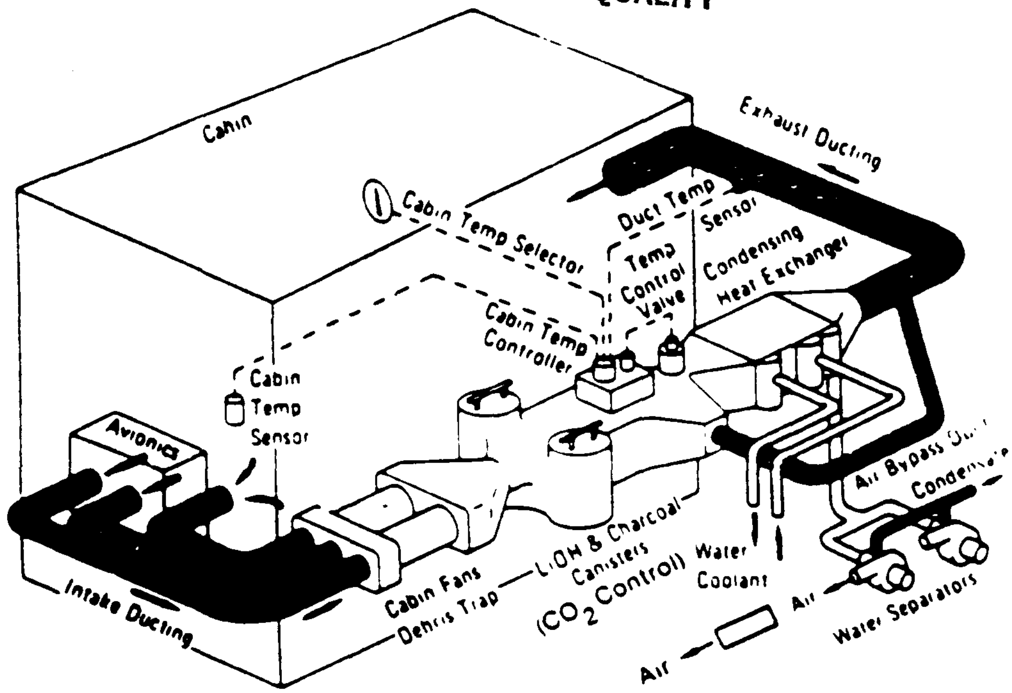

Figure 8-2: OTV Atmospheric Revitalization System

Only two canisters can fit into the system. One is actually used and the other is a reserve. When the components in the active canister are consumed, the system automatically switches to the reserve canister. Consequently, the canisters must be changed by the astronauts to insure system operation. Canister life, with three astronauts, will last 32 hours. Thus, 11 canisters will be needed for a 14 day mission. These will be stored above the system in one of the package compartments for quick and easy access.

The purified air then rejoins the main airflow. A temperature sensor in the crew and command module activates a valve that divides the air. A portion enters the air bypass duct where micro-organisms are filtered, and the other portion enters the cooling system. This cooling system is actually a condensing heat exchanger

that cools the air below the dew point. The heat build-up that occurs is reduced by the water cooling loop (this will be discussed in the next section). The air exits the heat exchanger and is then rejoined with the bypass air. Fresh oxygen is immediately added to the mixture from oxygen in the propulsion system, and the new mixture is vented into the crew and command modules. It is estimated that an air flow rate of 353 ft^3/min is needed to operate the system.

This system should maintain an air temperature between 55 °F to 70 °F and an air pressure of 14.7 psia. Nitrogen will be stored in separate tanks adjacent or across from the system so that the atmosphere will have a 20% O_2 and an 80% N_2 mix. A control in the crew module will permit desired selection of the temperature.

A repressurization airlock will be needed in the EVA module. This airlock is a cylinder whose diameter is 4 feet and whose height is 7 feet. It is placed on the outer edge of the EVA module. This airlock will facilitate crew exit and entrance into the OTV from the Space Station.

During EVA operations, the fully suited astronaut will enter the repressurization port or airlock from the command/crew module and seal the entrance door. Exit from the module may then be achieved accordingly. Upon completion of EVA, the astronaut re-enters the port, seals the exit hatch, and repressurizes the port. The air lock is repressurized by air that is bled from the command/crew module. It is estimated that the airlock will require about 6.65 lbm of air. This amount of air is not expected to effect the amount needed in the crew/command module, whose air requirements are about 202.1 lbm.

Additionally, the EVA module will not be pressurized at all, thus eliminating the need for a separate repressurization system. This will also reduce the amount of required O_2 and N_2. The astronauts will perform their necessary work in a vacuum environment. Entrance and exit into this portion of the module is made through a door in the airlock.

Since the OTV will be pressurized with and docked alongside the Space

Station, a full-scale repressurization system is unnecessary for the entire vehicle before mission operation. The OTV, before its severence with the docking bay, will activate its ECLSS. A safety factor of 1.25 has been included for the metabolic requirements of O_2 and N_2 to account for leaks in the system.

A monitor system will also be included to measure the oxygen, nitrogen, and carbon dioxide levels. This system will control oxygen and nitrogen supply and carbon dioxide removal. Information from the system will also alert the crew in case of malfunction. Table A1-2 in Appendix 1 gives the mass and power requirements to operate the complete air revitalization system. Most of the power will be needed to operate the ventilation system, the fans, and the condensing heat exchanger.

<u>Thermal Control</u>

A thermal control system is needed to remove excess heat away from the command and crew modules of the OTV. This excess heat originates from the electronic equipment on board, the fuel cells, the windows, and body heat from the astronauts.

This system is illustrated in Figure 8-3 and 8-4, and is comprised of a water and a Freon cooling loop. Water, cooled from the Freon interchanger, is routed to two heat exchangers. These heat exchangers cool the crew's drinking water. The water is then fed into the condensing heat exchanger (humidity control heat exchanger in the diagram) of the air revitalization system. The water passes into the inertial guidance heat exchangers which cool the guidance system of the OTV.

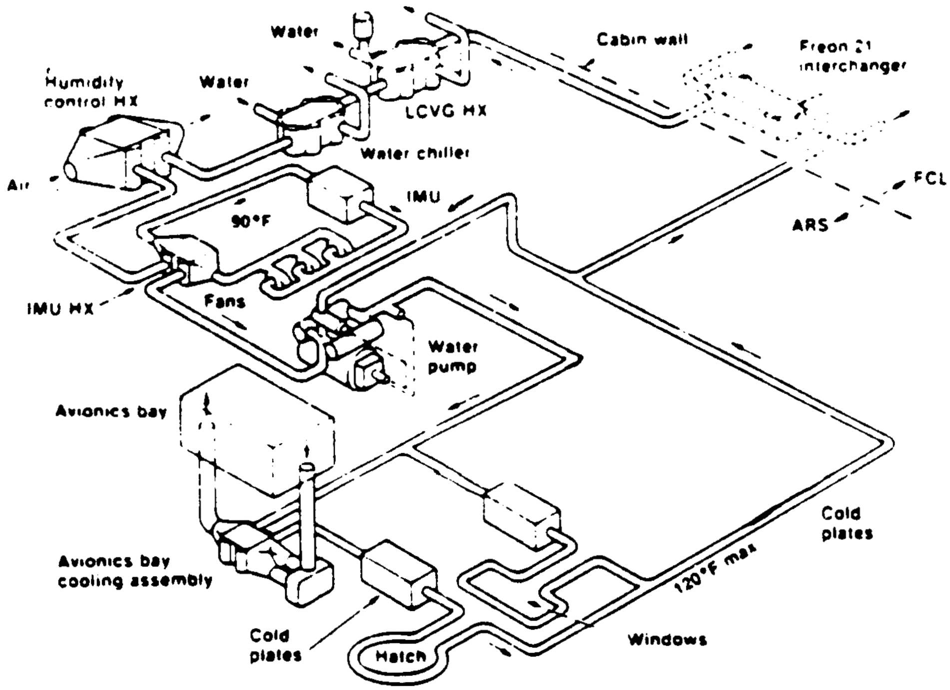

Figure 8-3: OTV Water Loop [1]

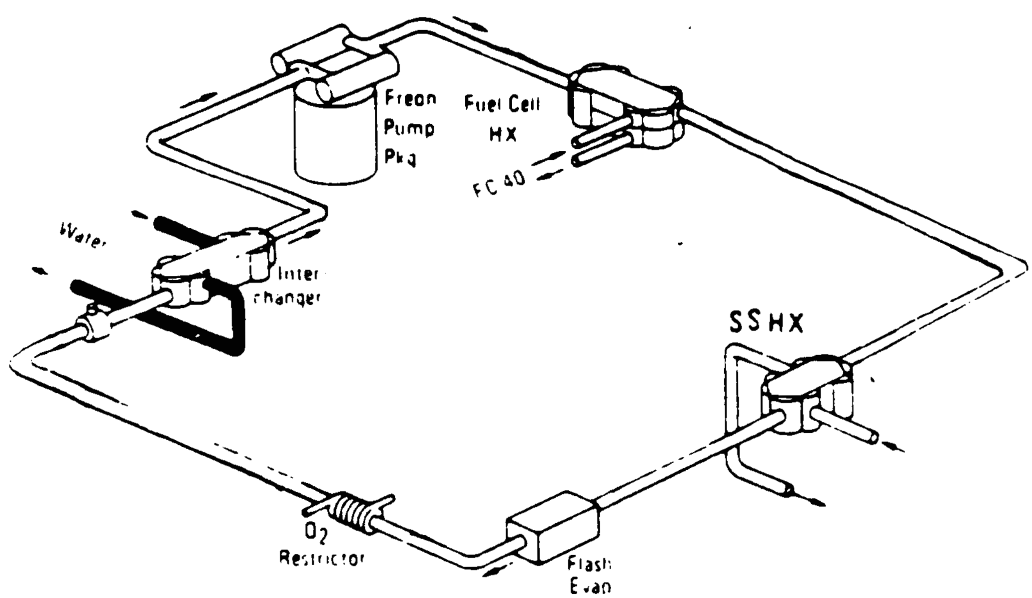

Figure 8-4: OTV Freon Loop [1]

Then, the partially heated water is routed to a water pump that returns a portion of the water back to the Freon interchanger and feeds another portion to both a cold plate and to the avionics bay cooling assembly which cools the avionics in the cockpit area. The water from this assembly goes into a cold plate where the water temperature is partially lowered. From here, the water is routed through window and hatch passages to cool these structures from sunlight and aerodynamic heating. The water then returns to the Freon interchanger. It has been estimated that a water flow rate of 221 lbm/hr is needed for adequate heat removal.

The Freon loop receives all the heat from the water cooling loop through an interface heat exchanger and cools the water to about 41 °F. A pump circulates the Freon as shown which flows to the fuel cell and power system heat exchangers. The flow rate must be at 780 lbm/hr for proper operation. These systems are Freon cooled accordingly in which the Freon now has been heated to 158 °F due to tremendous heat absorption.

The Freon then flows into the water flash evaporator where it is cooled to 38.8 °F. This evaporator vaporizes water to the outside of the vehicle and uses the heat of vaporization of the water to cool the Freon. The heated Freon is piped into a low pressure chamber through minute passages in the chamber walls. This pressure chamber is equipped with a vent to the outside. Water is then sprayed onto these walls where it evaporates, and this evaporation extracts the heat from the Freon. The extracted heat is later vented to the outside in the form of steam. The water that is needed for this operation should come as a by product from fuel cell operation.

After this process, the Freon is returned to the interchanger. Due to its toxic nature to humans, the Freon loop must be adequately sealed since it will be placed directly into the crew module. Sensors must be installed around this location to alert the crew of leaks. The mass and power requirements for the thermal control system can be found in Appendix Table A1-2. Power will be mainly needed to operate the pumps found throughout the water and Freon loops.

Crew Systems

The crew systems for the OTV will facilitate eating, drinking, sleeping, hygiene, and liquid/solid waste disposal. The requirements for this system are given in Appendix Table A1-2 Dehydrated and frozen foods will compose the main diet for the astronauts. At first, dehydrated foods were only considered, but owing to the rather long mission duration of 14 days, frozen foods were added for food variety. The dehydrated food is rehydrated by adding water (hot or cold, depending on preference) from the potable water system. The frozen foods are stored in a small freezer and prepared in a small microwave oven. Drinking water will be furnished from water produced from fuel cell operation, which will be cooled by the water- cooling loops before its actual use. Potable water can also be obtained from the condensation that forms from the cold plates in the thermal control system and from condensation that forms from the condensing heat exchanger in the air revitalization system. An emergency water storage tank will also be provided in case of system failure or malfunction.

Human wastes are handled with a toilet that separates the solid and liquid wastes which are placed into individual chambers by pressurized air. The solid wastes are stored until the OTV docks with the Space Station, where as the liquid waste (which also contains air odors) is injected into a separator. This device uses a rotating shell to force the liquid to the outer perimeter where it is removed and piped to the waste water tank for eventual ejection to outer space. The air odor is directed through a charcoal filter to remove the odors and then is returned to the cabins.

Hygiene will be provided through towel wipes laced with an antiseptic and compact shower bags like the ones found on the Space Shuttle. Water for these components will be taken from fuel cell operations. The water from the fuel cells will be at a temperature of 160 °F and will be maintained at this temperature until it is used to prevent the growth of bacteria. Prior to use, it will be cooled via the water cooling loop to about 110 °F.

The crew will use compact sleeping bags that will suspend freely from the sides of the crew module interior to sleep and rest. Three such bags will be included so that all crew members may sleep or rest simultaneously.

Radiation

Dose limits for radiation workers on earth are currently set at 5 Rem/year. Such limits are unrealistically low for astronauts [2]. Astronauts will be exposed to the danger of radiation unless they are protected with heavy radiation shielding. But space travel is a hazardous undertaking, and reducing the possibility of mission failure due to one type of hazard significantly below other types of hazards may be undesirable. Increasing the radiation shielding may in turn reduce the safety margin in propulsion or life support by adding too much weight, and may increase the overall risk of mission failure.

The amount of radiation that the astronauts of the OTV will receive during normal orbiting is negligibly small, even after 14 days. As seen in Tables 8-1 and 8-2, the OTV will receive 0.8 RAD per day (0.9 REM per day) at GEO and less than 0.1 RAD per day at LEO. Most danger comes from solar flares and the van Allen Belt. The time spent in the van Allen Belt on re-entry is very small, even with multiple pass entry. As shown in Table 8-3, Menees calculates that even for a 3 pass mission the OTV will graze the lower edge of the van Allen Belt only on the first pass, because the belt extends between 2.5 to 7 earth radii [3].

Solar flares, on the other hand, could cause significant radiation exposure. The protection that is afforded in the OTV is the structure of the OTV, the structure of the tanks, the propellant in the tanks, and the astronaut's space suits. The astronauts also have the option of turning the aerobrake to block radiation from solar flares if no pertinent operations are being performed at the time.

During our worst case mission, the OTV will receive only 7.2 REM, neglecting solar flares. As seen in Figure 8-5 and Table 8-4 this is a negligible amount and will not cause illness nor decrease the astronauts ability to perform a mission. Table 8-5 also demonstrates that the radiation will have an insignificant

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effect on the electronic components, lubricants, hydraulic fluids, glass, ceramic, and structural metals.

Table 8-1

SPACE RADIATION DOSE RATES (RAD/DAY) EXPECTED FOR ORBITAL MISSIONS[a]

Orbital altitude (km)	0.1 gm/cm²		1.0 gm/cm²		10 gm/cm²	
	Van Allen	Other[b]	Van Allen	Other	Van Allen	Other
300 Equator	3×10^2	<0.1	<0.1	<0.1	<0.1	<0.1
Polar	5×10^1	90	0.1	3	<0.1	0.2
400 Equator	2×10^3	<0.1	1	<0.1	0.3	<0.1
Polar	4×10^2	100	0.2	3	<0.1	0.2
600 Equator	1×10^4	<0.1	5	<0.1	2	<0.1
Polar	3×10^3	150	1	4	0.4	0.2
1000 Equator	1×10^5	<0.1	50	<0.1	15	<0.1
Polar	3×10^4	200	15	4	5	0.2
3000 Equator	3×10^5	<0.1	1000	<0.1	300	<0.1
Polar	1×10^5	300	300	5	100	0.3
10,000 Equator	1×10^6	<0.1	30	<0.1	10	<0.1
Polar	4×10^5	400	10	6	3	0.4
31,000 Equator	4×10^5	16	3	16	0.5	0.8
Polar	1×10^5	800	0.6	16	0.1	0.8

[a] All entries have 1σ limits of $\pm$ a factor of 3. Van Allen dose rates calculated for orbits in 1970, active Sun, assuming no more high altitude nuclear detonations. Galactic and flare doses calculated for solar maximum, 1% flare probability, averaged over 6 months.
[b] Other: includes flare and galactic radiation.

Table 8-2

SPACE RADIATION DOSE RATES (REM/DAY) EXPECTED FOR ORBITAL MISSIONS[a]

Orbital altitude (km)	0.1 gm/cm²		1.0 gm/cm²		10 gm/cm²	
	Van Allen	Other[b]	Van Allen	Other	Van Allen	Other
300 Equator	3×10^2	<0.1	<0.1	<0.1	<0.1	<0.1
Polar	5×10^1	250	<0.1	6	<0.1	0.2
400 Equator	2×10^3	<0.1	1.3	<0.1	0.3	<0.1
Polar	4×10^2	300	0.3	8	<0.1	0.2
600 Equator	1×10^4	<0.1	6.5	<0.1	2	<0.1
Polar	3×10^3	500	1.3	10	0.4	0.2
1000 Equator	1×10^5	<0.1	65	<0.1	16	<0.1
Polar	3×10^4	800	20	12	5	0.2
3000 Equator	3×10^5	<0.1	1300	<0.1	330	<0.1
Polar	1×10^5	1200	400	15	110	0.3
10,000 Equator	1×10^6	<0.1	35	<0.1	10	<0.1
Polar	4×10^5	2×10^3	12	18	3	0.4
31,000 Equator	4×10^5	50	3	50	0.5	0.9
Polar	1×10^5	4×10^3	0.6	50	0.1	0.9

[a] All entries have 1σ limits of $\pm$ a factor of 3. Van Allen dose rates calculated for orbits in 1970, active Sun, assuming no more high altitude nuclear detonations. Galactic and flare doses calculated for solar maximum, 1% flare probability, averaged over 6 months.
[b] Other: includes flare and galactic radiation.

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Table 8-3

Duration and apogee altitude for multipass aeroassist return missions

Number of atmospheric passes		GEO to Shuttle orbit Time, hr	GEO to Shuttle orbit Alt, km
1		6.1	400
2	#1	10.0	11,661
	#2		400
3	#1	14.1	16,773
	#2		7,670
	#3		400

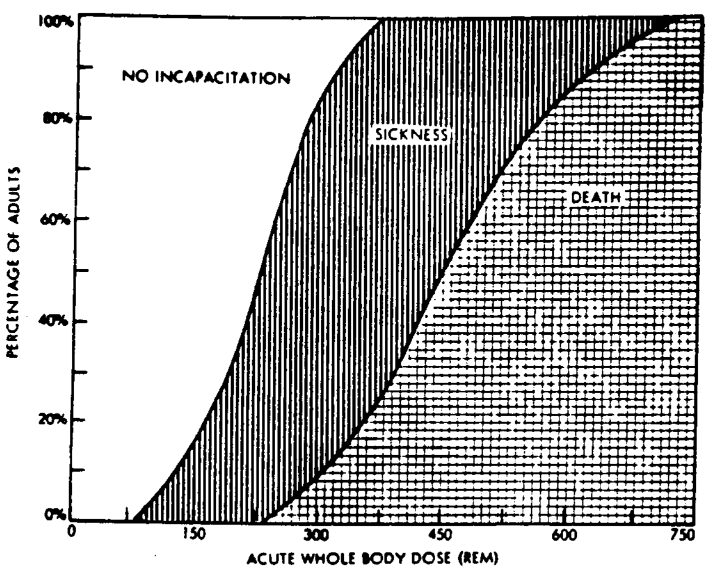

Figure 8-5: Incidence of sickness and death from acute radiation.

Table 8-4

EXPECTED EFFECTS OF ACUTE WHOLE-BODY RADIATION DOSES

Acute Dose (roentgens)	Probable effect
0–50	No obvious effect, except possibly minor blood changes
80–120	Vomiting and nausea for about 1 day in 5 to 10 percent of exposed personnel; fatigue, but no serious disability
130–170	Vomiting and nausea for about 1 day, followed by other symptoms of radiation sickness in about 25 percent of personnel; no deaths anticipated
180–220	Vomiting and nausea for about 1 day, followed by other symptoms of radiation sickness in about 50 percent of personnel; no deaths anticipated
270–330	Vomiting and nausea in nearly all personnel on first day, followed by other symptoms of radiation sickness; about 20 percent deaths within 2 to 6 weeks after exposure; survivors convalescent for about 3 months
400–500	Vomiting and nausea in all personnel on first day, followed by other symptoms of radiation sickness; about 50 percent deaths within 1 month; survivors convalescent for about 6 months
550–750	Vomiting and nausea in all personnel within 4 hours from exposure, followed by other symptoms of radiation sickness; up to 100 percent deaths; few survivors convalescent for about 6 months
1000	Vomiting and nausea in all personnel within 1 to 2 hours; probably no survivors from radiation sickness
5000	Incapacitation almost immediately; all personnel will be fatalities within 1 week

Table 8-5

RADIATION DAMAGE THRESHOLDS FOR CERTAIN CLASSES OF MATERIALS

Electronic components	10^1–10^3 rad
Polymeric materials	10^7–10^9 rad
Lubricants, hydraulic fluids	10^5–10^7 rad
Ceramic, glasses	10^6–10^8 rad
Structural metals, alloys	10^9–10^{11} rad

Chapter 9

Guidance, Navigation, and Control

The main purpose of the GNC system is to:

1. Determine the position of the vehicle.
2. Determine the magnitude, direction, and change of vehicle velocity.
3. Calculate and control manuveurs to reach specified position or rendezvous will a target satellite.

The position and the velocity of the OTV will be determined by information received from the planned Global Positioning System. This system, which will be composed of satellites positioned in 12-hour orbits, will produce signals that can be used to acurately determine vehicle position and velocity (time rate of change of position). Onboard, autonomous GNC will be provided by a combination of stellar tracker and laser-gyro inertial measurement units (IMU's). The stellar tracker is an opto-electrical device that is used to obtain vehicle attitute and position data from precise angular measurement of selected stars. The stellar tracker onboard the WWSR will have three axis imaging capability and a larger star catalogue than the Space Shuttle providing much higher accuracy and longer on-time [6]. The IMU provides vehicle attitude and velocity data from internal laser gyros and accelerometers. This part of the GNC system will play an important role during the aerobraking maneuver when the stellar tracker is unuseable and reception from the GPS system may be hampered by ionization of the air flow around the OTV.

The GNC system will be controlled by the general purpose computer systems. The computers will perform position and velocity determination from the various GNC sensors, will calculate needed maneuvers, and control the main engines and the attitude control system (ACS) to carry out the necessary changes.

Initially the WWSR OTV will be equiped with Ku-Band Rendezvous radar. This radar, which will also double as a communications link, will provide automatic

target detection and tracking to provide the range, velocity, roll, and pitch of a target satellite. This system will greatly reduce target location errors by allowing pre-rendezvous flight corrections. The Ku-Band radar can track a satellite with an active transponder at a range of 400 miles and a dead satellite at a distance of 14 miles [1].

Reaction Control System (RCS)

The Attitude Control System will respond to flight software commands and GNC inputs via the Data Management System to control the OTV's attitude, trajectory, rendezvous maneuvers. The ACS jets will use N_2H_4 hydrazine fuel and will each produce a thrust of 111 Newtons at a specifc impulse of 220 seconds [5]. There will be a total of 36 jets arranged in 8 locations to provide complete translational and rotational control of the OTV during rendezvous, docking, trajectory correction and aerobraking. Four stations, each with four thrusters, are located around the EVA module. A tank within the EVA module supplies the fuel for these four stations. The remaining four stations are attached along the rim of the aerobrake. These stations have five thrusters each, with some firing through the edge of the brake itself. Each of the stations has its own hydrazine fuel tank. The OTV will carry a maximum of 2900 lbs of hydrazine fuel.

Chapter 10

Data Management System

The Data Management System (DPS) will control and monitor the OTV during the course of each mission. Some of these functions include:

1. Support of GNC system, including calculation and control of vehicle position and trajectory.
2. Monitoring and control of vehicle subsystems including electrical power, environmental control, and main engines.
3. Processing vehicle data for radio transmission and responding to uplinked vehicle commands.

The DPS systems will consist of 3 onboard computers, 2 mass memory units, 3 crew input/output stations, and the data bus network. The onboard computers will be IBM's new 1750A (Air Force Standard) avionics system [1]. These high speed, high capacity machines were choosen because of the enormous computing power needed during the aerobraking maneuver. The IBM system provides the highest computing speed in the smallest box. The mass memory units will be write-once optical discs. Each of the two memory units will contain copies of the flight software and star catalogue for the stellar tracker and will provide memory for mission data storage.

The forward flight deck will consist of three flat screen plasma displays, two keypads, and the numerous controls and switches that operate all of the subsystems of the OTV. All phases of operation of the OTV are controlled from the flight deck, either automatically though the computers or manually. The remaining display and keyboard, attached within the avionics component compartment, can be used as a work station off the flight deck.

The data bus network provides a means of communication between each of the vehicles subsystems and the DMS. The data buses will be high density optical

cable to reduce weight, size, and electromagnetic interference. The multiplexer/demultiplexer systems will convert DMS and subsystem signals to coded light signals for transmission over the data bus network. The data bus and multiplexer systems will be tripley redundant [4].

Chapter 11

Communication System

The communication system provides direct voice and data links between the OTV and the space station, ground control, and EVA astronauts. The communications system onboard will consist of S-Band, Ku-Band, and UHF radio frequency links. The S-Band phase modulation system will be used to transmit and receive voice and data to and from the space station and ground control. The system can either be used in direct link mode or relayed through the Tracking Data Relay Satellites (TDRS). The S-Band system will be redundant (two independent systems) as it is the most versitile and important communications link. The Ku-Band system (same device as rendezvous radar) will be used to transfer data at rates much higher than the S-Band system. The Ku-Band system can only relay data through TDRS and is not operational during aerobraking (antenna will be stowed) or when being used as rendezvous radar. The UHF system will be used for voice communication between the OTV and EVA astronauts and during docking procedures with the space station [3]. The entire communications system will be interfaced with the Data Management System to control reception, transmition, command execution and data telemetry.

The antennas for the S-Band and UHF radios will be flush mounted on the structure of the OTV. Four sets of redundant S-Band antennas, spaced at 90 degree intervals around the EVA module, will provide complete transmission and reception coverage with the space station and ground control either directly or through TDRS. Three UHF strip antennas, one near the docking berth, one inside the command module, and one inside the EVA capsule, will provide communications with and between astronauts before and during EVA and with the space station during docking. In addition, small headset radios can be used inside the command module to allow all of the astronauts to communicate with each other as well as be linked into the entire comm net.

The Ku-Band intergrated radar and communications system antenna is a

deployable 3-foot parabolic dish [1]. It will be stowed down along the side of the EVA module during the aerobraking maneuvers to protect it from drag forces and aerodynamic heating.

Chapter 12

Satellite Repair and Recovery System

The satellite repair and recovery system is designed to satisy the objectives of the mission - to attempt a repair (or refueling) of a dysfunctional geostationary satellite and, if unsuccessful, dock with the satellite and return it to the space station at LEO for further servicing. This system will reduce the costs of satellite operation. As the cost of replacing a satellite far exceeds the cost of a repair mission, significant savings can be gained. These savings are evidenced by past repair missions [2].

Table 12-1: Satellite Repair Missions

Satellite	Estimated Cost	Repair Mission	Comments
Palapa	200 million	10 million	Resold for 60 million
Solar Max	270 million	43 million	Redeployed

The satellite recovery and repair system consists of 6 items:

1. Manned Maneuvering Unit (MMU) - see Figure 12-1
2. Extravehicular Mobility Unit (EMU) - see Figure 12-2
3. Manipulator arm
4. Grappling device
5. Repair Tools
6. Docking Ring

The above items function collectively to create an integrated system for repairing or recovering the satellite. The following typical mission employing the system serves to describe the characteristics and functions of each component.

Upon rendezvous of the OTV with the satellite to within 150 feet by means of RCS, an astronaut dons the EMU and enters the EVA module through the passage provided. Although the suit is heavy (approximately 300 lbs.) the zero gravity environment allows relatively easy manipulation of the EMU. The astronaut then proceeds to put on the externally mounted MMU, which is also stored in the EVA chamber. The escape hatch to the outside is sufficiently large to allow the astronaut to move away from the OTV without complication due to the relatively awkward MMU.

The MMU is a self propelled backpack device for maneuvering the astronaut through space as a free flier. The MMU is equipped with twenty-four fixed gaseous nitrogen thrusters each capable of delivering of delivering 1.7 lbs. of thrust and allowing six degree of freedom maneuverability. Additionally, the MMU is equipped with an automatic altitude hold which provide sufficient control to damp out the motion induced by the movement of the astronaut's limbs. It is designed to be failsafe - fully redundant controls in electrical, electronic and propulsion subsystems. Electrical power is supplied by two batteries, each with an energy capacity of 752 watt-hours. The dimensions of the MMU are approximately 50 in. high, 33.3 in. wide, 27 in. deep with arms in launch position and 48 in. deep with arms in the extended flight position. The 340 lb. unit fully charged with 26 lbs. of propellant can function for a six hour EVA and has a range of 3000 ft. At full charge, the two aluminum pressure tanks with Kevlar overwrap (pressurized to 3000 psia) can induce a propulsive delta v of 66 fps to the 800 lb. combination of man, MMU and EMU. This device has performed flawlessly on three previous misssions and has proven its goal to move an astronaut easily, accurately, and reliably in free flight [1].

The MMU configuration proceeds to the disabled satellite and matches angular velocity. At this point, several options are presented. The astronaut may attempt to repair the satellite by attaching the MMU to the satellite by a means determined by the specific satellite. Previous missions such as that to repair the Solar Maximum Mission Satellite used a device known as a trunion pin adapter (see Figure 12-3) to make this attachment. Simple operations such as replacing a

satellite module may be accomplished in this fashion. More likely however, the satellite may require more sophisticated servicing. Therefore, the astroaut will need to prepare the satellite for return to the OTV by means of the manipulator arm. This necessitates the use of grappling device attached to the satellite to which the manipulator arm may secure itself. Unfortunately, there are, presently, no universal grappling devices for satellite repair. An optimum solution to this problem would be the standardization of all future satellites (see Figure 12-4) to promote easy repairability. Then a universal "stinger" device such as that used to retrieve the Westar VI satellite (see Figure 12-5) may be connected to the satellite and the astronaut-MMU configuration could propel the satellite to within reach of the OTV's manipulator arm.

Without this optimum satellite standardization, however, a number of alternatives arise to continue the mission. Instead of using the MMU to propel the satellite to the OTV (which can only be accomplished reliably if the mass of the satellite is sufficiently low), the MMU may be used to attach a device to to the satellite to which the manipulator arm may attach itself. By maneuvering the OTV to within 15 feet of the satellite the manipulator arm may be employed to its greatest potential.

The manipulator arm of Figure 12-6 may be used to grasp the satellite and lower it to the docking berth on the outside of the EVA module. The ability of the docking berth to be adapted to properly fit and securely hold the satellite is essential and unfortunately, subject to the same limitations of the grappling device described above. Once this problem has been overcome, however, the manipulator arm assumes another role as a "cherry picker" [5]. To this extent, the arm serves to maneuver an astronaut around the satellite for further satellite servicing. As visibility from within the OTV is limited, the manipulator arm is teleoperated by a camera mounted just behind the end of the grasping arm. This arm will require six degrees of freedom to successfully attach to the satellite and permit approach from various angles.

The above components compromise the satellite repair and recovery system of

the WWSR orbital transfer vehicle. As there are great variations in current satellite design, the non- rigidity of proposed system is obvious. Modifications may be necessary as dictated by the individual mission. The ability of the space station to stock a sufficient supply of repair and recovery system components is essential to the functioning of the OTV.

The MMU is a self-contained backpack for propelling an astronaut during EVA. Twenty-four fixed gaseous nitrogen thrusters, each delivering 1.7 lb of thrust, allow six degrees of freedom maneuvering ability.

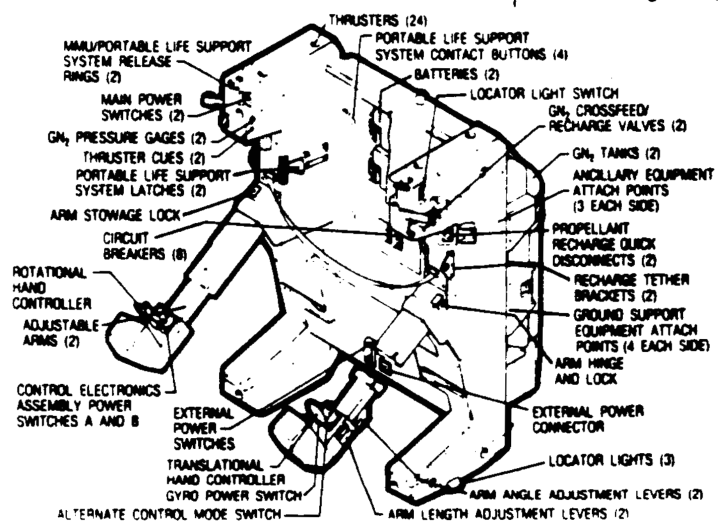

Figure 12-1: The Manned-Manuevering Unit

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by Andrew J. Hoffman

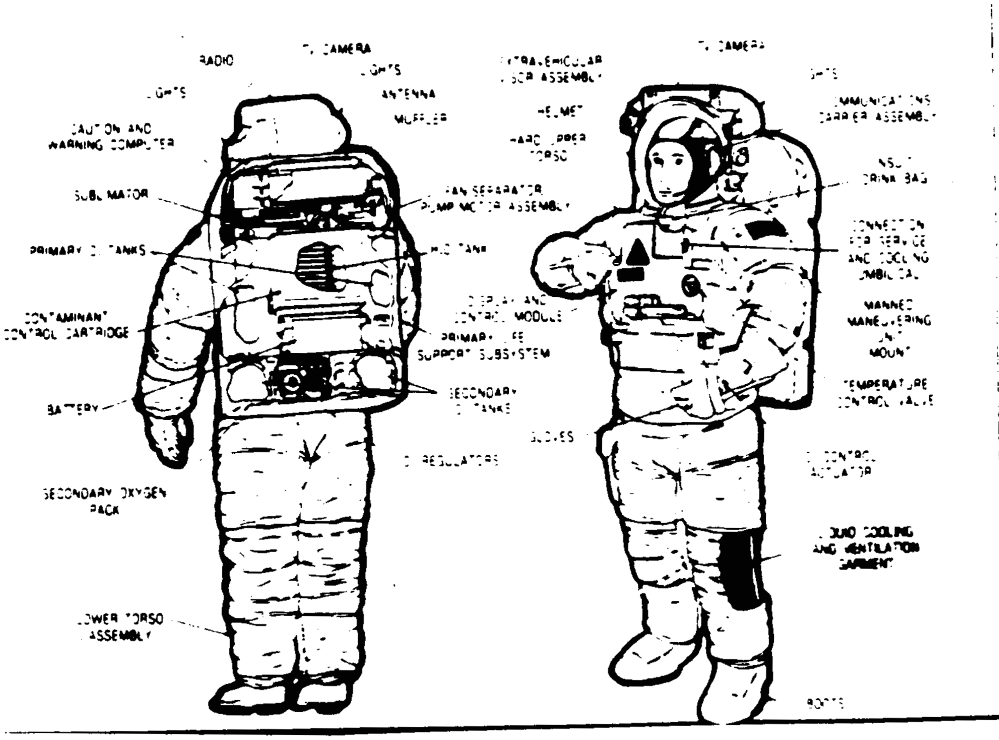

Figure 12-2: The Extravehicular Mobility Unit

Figure 12-3: Trunion Pin Attachment Device

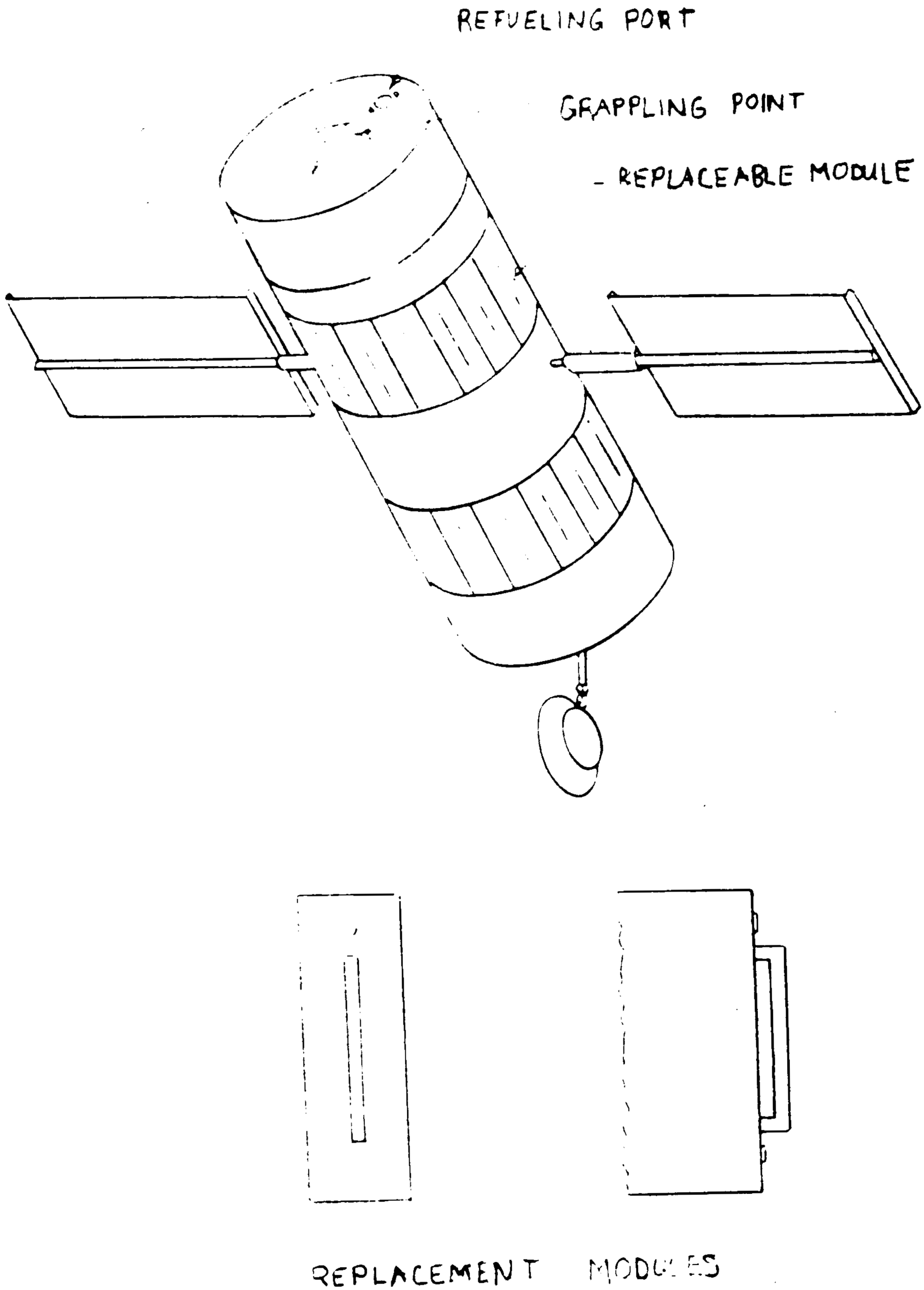

Figure 12-4: Serviceable Satellite Configuration

Figure 12-5: "Stinger" Device in Use

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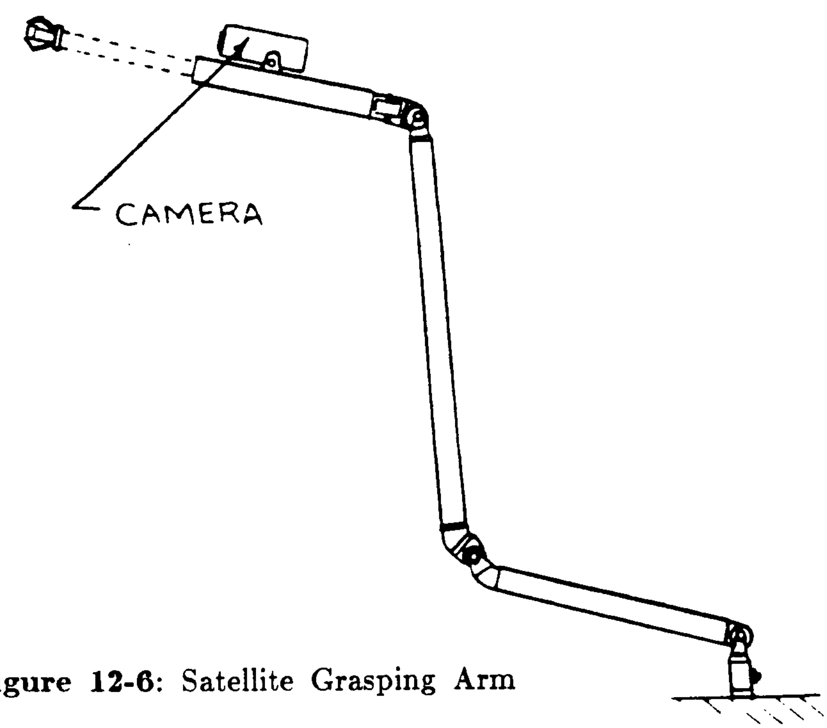


Figure 12-6: Satellite Grasping Arm

Chapter 13

Cost Analysis

The contract on any new type of design is inevitably determined, in part, as a function of its cost. Although the RFP for this project never specified a requirement to consider cost efficiency, the WWSR project team sought to integrate the most cost effective, usable components whenever possible. To this extent, many tried and tested devices are included in the design to avoid the comparatively large costs associated with research and development of state-of-the-art technology. Nevertheless, these costs could not be eliminated in all instances and were responsible for a large portion of the overall cost. The cost of the computer hardware and software necessary to successfully complete the complex aerobraking maneuver, for example, compromised nearly 13% of the overall cost of the OTV.

However, the seemingly large expenditure on these computer systems is justified by the argument that the proposed aerobrake configuration will produce a dollar savings of over fifty percent as compared to existing orbital transfer vehicle concepts using all-propulsive methods of transfer between LEO and GEO. In approximately ten "typical" missions, this savings will compensate for the undeniably large research, development and systems testing costs which necessarily accompany the installment of any new technology.

The approximate costs of the majority of the systems, structures, and components are provided on the following page. Wherever possible, the costs of previously used components were researched and economically scaled to determine the current figure. In some instances, such as the determination of the computer software cost and the aerobrake research and development cost, some fundamental concepts of engineering cost estimation and analysis were employed to determine a numerical figure.

The cost breakdown on the following page does not include the cost of the fuel itself or the cost of transporting the OTV or the fuel which it requires to the

space station. One should not assume, however, that these costs are negligible. In fact, current NASA figures indicate that the cost of transporting a mass aboard the shuttle to low earth orbit is approximately $1500/lb. Based on the overall "dry" weight of the OTV, we can expect an additional expense of nearly $40 million dollars just to transport the OTV in pieces to its berth at the space station. In order to fully fuel the OTV for the worst possible case would require yet another expense of approximately $70 million. This figure is based on the contention that the fuel is put in orbit by a more cost effective means than in the cargo bay of the space shuttle.

The results of the cost analysis are open to a variety of interpretations. The final cost of development of a single OTV was determined to be $850 million ($970 million including transportation costs). In light of the fact that the modern version of the shuttle costs approximately $1 billion and space station cost projections waver around $9 billion, we can conclude that the cost of this project is, by no means, insignificant. Nor can any realistic cost decrement for the specified design parameters be expected. This is not to say that such a design project should be abandoned. The 3-man crew capability provides great opportunity for the repair of malfunctioning or dead satellites. However, some alterations of the design and/or mission specifications are very appropriate. To this extent, the man-rated functioning, coupled with the capability of the OTV to deliver and/or recover a 30,000 lbm object from LEO and GEO impose significant weight additions to the mission which, consequently, boost both mission and design costs tremendously. Therefore, it is the recommendation of WWSR, Inc. to modify the mission requirements. The manned OTV will be of great value to the satellite repair function of the design. However, when considering the satellite deploy and recovery function of the design, consideration of other options such as an unmanned OMV may prove to be more cost effective.

C - 2

Table 13-1

Numerical Breakdown of Project Orion Costs

Item	Cost (In Millions of Dollars)
Aerobrake	150
Fuel Tanks	20
6 liquid O_2	
6 liquid H_2	
Avionics	
Hardware	20
Software	130
Pratt & Whitney Engines (2)	50
Power Generator	15
2 United Technology Fuel Cells	
Battery	
EVA Module (with docking mechanism)	25
Reaction Control System (RCS)	5
Satellite Recovery System	5
Manipulator Arm & Grappling Device	
MMU & EMU	
Berthing Device	
Tools	
Main Cabin Structure and Components	200
Pressurization and Temperature Control System	45
Program Development and Management	75
Research, Development, and System Testing	250
Unaccounted Incidentals	75
Summation of Costs	1050

Chapter 14
Managing Project Orion

This section is a brief discussion of how Project Orion should be managed. Managing Project Orion will be a joint effort between WWSR and NASA. WWSR will be responsible for establishing the contractors and subcontractors of the project as well as monitoring the work of these corporations. NASA in turn will monitor WWSR as well as manage the deployment of the OTV. In monitoring WWSR, NASA's responsibilities will consist of approving the decisions, selections, and funding of our corporation. NASA will have the power to override decisions made by WWSR. WWSR's responsibilities with respect to its contractors and subcontractors will be similiar to that of NASA's. WWSR will be responsible for the distribution of funding from NASA to the contractors as well as approving major decisions and designs developed by the contrators. It is expected that the relationships between contractors and their subcontractors will be managed in a similiar fashion.

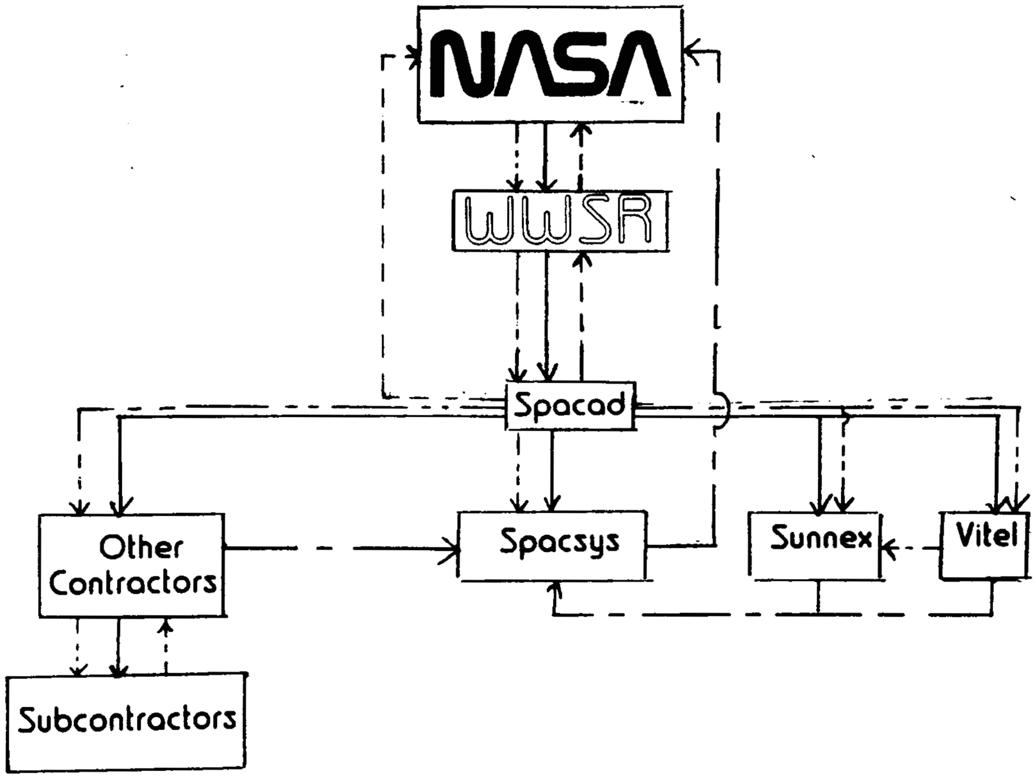
WWSR itself is a relatively new corporation in the space market. We are, however, one of the oldest airframe manufacturers in the country and have enjoyed a very successful partnership with the government in ensuring the defense of this country. Fifteen years ago, WWSR went through a major restructuring to assure viability into the twenty-first century. It was decided then that WWSR would continue its work on development of civilian and military aircraft as well as devote a substantial amount of capital into research and development of space systems - an area we felt confident would provide us with many exciting and challenging projects. Our goal was to be prepared to make a bid on a major space contract in ten years. WWSR then began to merge and aquire several firms active in developing space systems. WWSR Inc. is now divided into six fairly automonous "companies": WWSR Aircraft, Sunnex Controls, Airprop Engines, Vitel Electronics, WWSR Space Systems (Spacsys), WWSR Space Analysis Division (Spacad).

The development of the OTV in this proposal was primarily the responsibility of Spacad with appropriate input from Spacsys, Sunnex, and Vitel. Spacad will be

the company responsible for monitoring the work on Project Orion. Spacsys will manufacture the crew module, EVAM, propellant tanks, and support structure. This will be done at our recently converted airframe facilities in California. Sunnex will be responsible for developing and manufacturing the electronic and mechanical control systems for the OTV. Vitel will manufacture most of the electonic components needed by the other companies. Some of the systems and components of the OTV that will be contracted out will be the main engines, thermal controls, communication systems, RCS engines, and the aerobrake and thermal tiles. All components not directly manufactured by Spacsys will be installed at the company's plant. Figure 14-1 illustrates Project Orion's manufacturing and management structure.

Once the completed system is delivered to NASA, WWSR's responsibilities will be to provide replacement components for the OTV and to consult NASA through Spacad in mission planning. It is Spacad's opinion that NASA should employ the same system of management for Project Orion that it proposes to use for managing the Space Station. Assuming NASA uses the management system proposed by Granville Paules [2], Project Orion will be a subsystem of Space Systems Operations. Space Systems Operations controls space system activities concerning the Space Station that occur in orbit or on the ground. The subsystem which will monitor Project Orion will consist of six divisions: User Operations Support, Mission Planning, Predeployment/Postdeployment Operations, Integrated Logistics Support, Market Research, and Cost and Financial Managment. Each of these divisions will consist of members from NASA, WWSR, and users. User Operations Support will be responsible for assisting users in planning and directing the allocation of the OTV. Mission Planning will create the optimal strategy for deployment of missions set up by User Operations. Predeployment/Post-deployment Operations will manage the functions of final servicing, integration, and processing of subsystems just before and after the OTV leaves and returns to the Space Station. Integrated Logistics Support will delegate the logistic requirements of the various users. Market Research will serve as a catalyst for developing new areas in which the OTV can be employed. Cost and Financial Mangament will promote cost-effective operations.

PROJECT ORION



Management ———

Consulting – – –

Funding —·· —

Finished Product — - ——

Figure 14-1: Management and Manufacturing Structure

Chapter 15
Mission Planning

The purpose of this section is to present three scenarios for possible missions for WWSR's OTV. The description of the missions include mission objective, OTV configuration and weight estimates, fuel requirements, time of various actions, and delta v's and fuel consumed for various manuevers.

Mission A: Worst Case Scenario

Mission Objectives: The OTV will leave the Space Station carrying components for constructing a platform at GEO (payload, 24000 lbm). The OTV will also carry provisions for a full crew of 3 for a 14 day mission. Eight days on station will be anticipated for construction of the platform. Upon completion of construction, the OTV will returned unloaded to the Space Station.

Configuration: 6 pairs of propellant tanks, 2 MMUs, 3 crew.

Weight Estimates:

System	Weight (lbm)
ECLSS	3560
Tanks and Supporting Structure (6 pairs)	3660
Engine System	1050
Crew Cabin, EVA, and Components	13260
Aerobrake	2800
Electronics	985
EPS	2215
RCS	3350
MMU (2)	1280
Crew (3)	510
Total (Dry)	32670
Payload (Out)	24000
Payload (Return)	0
Total Propellant	125000

Mission Profile: The mission profile for the delivery and setup of a 24,000 lbs space platform to geosynchronous orbit is shown in Table 15-1. Following separation from the Space Station and subsequent systems checkout, the OTV performs a phasing orbit injection burn (PIB). The phasing orbit is designed to bring the

OTV to the transfer orbit injection point at the proper time so it will arrive at the correct location in GEO. The transfer injection burn places the OTV in a Hohmann elliptical transfer to GEO, which lasts approximately five hours. Following circularization at GEO, the OTV can remain on station for eight days to deploy (i.e. possibly construct) the space platform.

After deployment is completed, an injection burn places the OTV in a GEO-LEO transfer orbit that will take it through the Earth's atmosphere. The first aerobraking pass, dipping the OTV to a height of 85 kilometers above the Earth, lasts only five minutes and leaves the vehicle in an intermediate orbit. Based on the results of the first pass, correction burns take the OTV through the atmosphere a second time. This time the maneuver lasts about 11 minutes and places the OTV in an orbit that can be circularized at LEO by a small propulsive burn. Note that the main fuel tanks are not full to capacity and that there is still fuel in reserve. This indicates that the OTV could carry even heavier payloads than 24,000 lbm.

Table 15-1

Profile of Mission A: GEO Delivery of 24,000 lbm Payload

Event	Duration (hrs)	ΔV (m/s)	Prop. (lbm)	
Separate	4.0	3	251	(RCS)
Phase Injection	0.2	1400	44793	
Coast	3.0	5	315	(RCS)
Transfer Burn	0.1	1006	25112	
Coast & Correct	5.0	10	512	(RCS)
GEO Circularization	0.1	1826	34168	
Trim	12.0	5	176	(RCS)
Deliver Payload	196.0	10	240	(RCS)
Phase	10.0	-	-	
Transfer Burn	0.1	1845	16155	
Coast & Correct	5.0	10	164	(RCS)
Aerobrake Manuever	0.1	10	164	(RCS)
Coast	3.2	5	81	(RCS)
Aerobrake Manuever	0.2	10	162	(RCS)
LEO Circularization	0.1	200	1392	
Rendezvous & Dock	6.0	20	310	(RCS)

Launch Mass:	181,270 lbm	
Return Mass:	33,798 lbm	
Total Elapsed Mission Time:	240 hrs	
Total H_2-O_2 Prop. Used:	121,616 lbm	(3384 lbm reserve)
Total RCS Fuel Used:	2,375 lbm	(525 lbm reserve)

Mission B: Satellite Repair

Mission Objectives: The OTV will leave the Space Station travelling to GEO and carrying no payload. The OTV will also carry provisions for a crew of 2 for a 6 day mission. At GEO, the crew will service two satellites. It will be anticipated that servicing will take one day for each satellite. Upon completing service of the first satellite, the OTV will make a epoch change of 30° to rendezvous with the second satellite. Upon completing service of the second satellite, the OTV will returned unloaded to the Space Station.

Configuration: 4 pairs of propellant tanks, 2 MMUs, 2 crew.

Weight Estimates:

System	Weight (lbm)
ECLSS	2335
Tanks and Supporting Structure (4 pairs)	2440
Engine System	1050
Crew Cabin, EVA, and Components	13260
Aerobrake	2800
Electronics	985
EPS	1615
RCS	3250
MMU (2)	1280
Crew (2)	340
Total (Dry)	29445
Payload (Out)	0
Payload (Return)	0
Total Propellant	88000

Mission Profile: The profile for a mission to service two geosynchronous satellites is shown in Table 15-2. After the OTV is fitted with three fueled tanks, it separates from the space station and preforms full systems checks. The OTV then uses the same sequence of phase injection and transfer orbit injection burns to arrive at the proper location in GEO as detailed for Mission A. At GEO the RCS engines are used to maneuver the OTV to retrieve the first satellite. Depending on its configuration, the satellite may be recovered using either the robot arm or with the assistance of an astronaut in a Manned Maneuvering Unit (MMU). The satellite is berthed to the OTV where the EVA astronauts can effect repairs. The robot arm is particularly useful for moving an astronaut around the satellite, providing a mobile work platform. After repairs are completed, the satellite can be deployed and fully tested to assure proper operation before the OTV moves to the next satellite.

To change its placement in GEO, the OTV performs an epoch change burn which places the vehicle in an orbit slightly smaller (and more elliptic) than GEO. This brings the OTV back to GEO, the epoch change orbit's apogee, in 21.6 hours (less than the 24 hour period of GEO. By recircularizing, this effectively moves the OTV forward by about 30° in GEO.

The same procedure outlined above is used to recover and repair the second satellite. After completing the second round of repairs, the OTV will perform a transfer orbit injection burn which will take it through the Earth's atmosphere twice and return it to LEO.

Table 15-2

Mission B Profile: GEO Servicing of Two Satellites Separated by 30°

Event	Duration (hrs)	ΔV (m/s)	Prop. (lbm)	
Separate	4.0	3	163	(RCS)
Phase Injection	0.2	1400	28925	
Coast	3.0	5	203	(RCS)
Transfer Burn	0.1	1006	16216	
Coast & Correct	5.0	10	331	(RCS)
GEO Circularization	0.1	1826	22063	
Rendezvous	6.0	25	566	(RCS)
Repair	24.0	-	-	
Unload Payload	3.0	5	112	(RCS)
Epoch Change Burn	0.1	200	1929	
Coast	21.6	5	107	(RCS)
GEO Circularization	0.1	200	1848	
Rendezvous	6.0	25	513	(RCS)
Repair	24.0	-	-	
Unload Payload	3.0	5	101	(RCS)
Transfer Burn	0.1	1845	13740	
Coast & Correct	5.0	10	140	(RCS)
Aerobrake Manuever	0.1	10	139	(RCS)
Coast	3.2	5	69	(RCS)
Aerobrake Manuever	0.2	10	138	(RCS)
LEO Circularization	0.1	200	1184	
Rendezvous & Dock	6.0	20	263	(RCS)

Launch Mass:	117,455 lbm	
Return Mass:	28,300 lbm	
Total Elapsed Mission Time:	115 hrs	
Total H_2-O_2 Prop. Used:	85,907 lbm	(2093 lbm reserve)
Total RCS Fuel Used:	2,846 lbm	(54 lbm reserve)

Mission C: 15,000 lbm Payload Up and Back

Mission Objectives: This mission is used to compare the performance of WWSR's OTV to that of MOVERS'. Essentially, the mission consists of carrying a payload of 15,000 lbm from the Space Station to GEO and back. This payload might be some sort of experiment assembly used for SDI testing. The OTV will carry a crew of 3 for a total mission time of 7 days.

Configuration: 6 pairs of propellant tanks, 2 MMUs, 3 crew.

Weight Estimates:

System	Weight (lbm)
ECLSS	2500
Tanks and Supporting Structure (6 pairs)	3660
Engine System	1050
Crew Cabin, EVA, and Components	13260
Aerobrake	2800
Electronics	985
EPS	1730
RCS	3350
MMU (2)	1280
Crew (3)	510
Total (Dry)	31125
Payload (Out)	15000
Payload (Return)	15000
Total Propellant	132000

Mission Profile: The mission profile for the delivery to GEO and return to the Space Station of a 15,000 lbs payload is shown in Table 15-3. Following separation from the Space Station and subsequent systems checkout, the OTV performs a PIB. The transfer injection burn places the OTV in a Hohmann elliptical transfer to GEO, which lasts approximately five hours. Following circularization at GEO, the OTV can remain on station for five days to perform the necessary experiments.

After completing the experiments, the OTV will return to the Space Station with the payload following similar procedures for returning to LEO as described in the Mission A profile.

Table 15-3

Mission C Profile: 15,000 lbm Payload Up and Back

Event	Duration (hrs)	ΔV (m/s)	Prop. (lbm)
Separate	4.0	3	247 (RCS)
Phase Injection	0.2	1400	44015
Coast	3.0	5	310 (RCS)
Transfer Burn	0.1	1006	24675
Coast & Correct	5.0	10	503 (RCS)
GEO Circularization	0.1	1826	33575
Trim	12.0	5	173 (RCS)
Station Keeping	120.0	10	345 (RCS)
Phase	10.0	-	-
Transfer Burn	0.1	1845	23210
Coast & Correct	5.0	10	236 (RCS)
Aerobrake Manuever	0.1	10	235 (RCS)
Coast	3.2	5	117 (RCS)
Aerobrake Manuever	0.2	10	233 (RCS)
LEO Circularization	0.1	200	2000
Rendezvous & Dock	6.0	20	445 (RCS)

Launch Mass:	178125 lbm	
Return Mass:	47803 lbm	
Total Elapsed Mission Time:	168 hrs	
Total H_2-O_2 Prop. Used:	127474 lbm	(4526 lbm reserve)
Total RCS Fuel Used:	2844 lbm	(56 lbm reserve)

Conclusion

WWSR has presented what it feels is the most optimal design for a chemical propellant, manned OTV that fulfills the previously described constraints. Even though this is the end of our report, we feel that much more research can be done. WWSR's OTV is designed to be versatile and modular. Many more missions other than the ones described in this proposal may be possible with minor design or component changes. We especially feel confident that with a small amount of development, our OTV would be capable of performing missions to the Moon. This could include orbiting to retrieve payloads or landing on the lunar surface. Because of its modular design, WWSR's OTV will truly be the orbital transfer vehicle for the 21st century.

References

Introduction

1. *Pioneering the Space Frontier: The Report of the National Commission on Space.* New York: Bantam Books, 1986.

Chapter 2

1. Bragg, B.J. *A Design Study for an Aeroassist Flight Experiment.* Houston, Texas: NASA, June 1985.

2. Dotts, R. L. "Shuttle Orbiter Reusable Surface Insulation Thermal Performance, Entry Vehicle and Thermal Protection Systems: Space Shuttle, Solar Starprobe, Jupiter Galileo Probe." *Progress in Astronautics and Aeronautics* Vol. 85, 1983.

3. Gamble, J.D. *Lateral Directional Requirements for a Low L/D Aeromaneuvering Orbital Transfer Vehicle.* AIAA Paper 84-2123, August 1984.

4. Hollo, Steven D. *OTV2.* Computer Simulation of Aerobraking Maneuver. Unpublished, 1988.

5. Li, C.P. *Computation of Three-Dimensional Flow about Aerobrake Configuration.* AIAA Paper 86-0566, January 1986.

6. Meness, G.P. "Design and Performance of a Conical Aerobrake Orbital Transfer Vehicle." *Thermal Design of Aeroassisted Orbital Transfer Vehicles.* H.F. Nelson ed. New York: AIAA, 1984.

7. Meness, G.P. "Determining Atmospheric Density Using a Space-Launched Projectile." *Journal of Spacecraft and Rockets* Vol. 23 No. 3, May-June 1986.

8. Meness, G.P. "Thermal Protection Requirements for Near-Earth Aeroassisted Orbital Transfer Vehicle Missions." *Thermal Design of Aeroassisted Orbital Transfer Vehicles.* H.F. Nelson ed. New York: AIAA, 1984.

9. Park, C. *A Survey of Aerobraking Orbial Transfer Vehicle Design Concepts.* AIAA Paper, January 1987.

10. Rehder, John J. "Multiple Pass Trajectories for an Aeroassisted Orbital Transfer Vehicle." *Thermal Design of Aeroassisted Orbital Transfer Vehicles.* H.F. Nelson ed. New York: AIAA, 1984.

11. Shih P.K. "Low Lift-to-Drag Aerobrake Heat Transfer Test at Mach 10." *Thermal Design of Aeroassisted Orbital Transfer Vehicles.* H.F. Nelson ed. New York: AIAA, 1984.

12. Walberg, Gerald D. *A Review of Aeroassisted Orbit Transfer.* AIAA Paper 82-1378. AIAA 9th Atmospheric Flight Mechanics Conference, San Diego, August 1982.

Chapter 3

1. Brown, J.R. *System Requirements Impact on Advanced OTV Engine Design.* AIAA Paper 85-1340, July 1985.

2. Charhut, D.E. and W.J. Ketchut. "Future Requirements and Applications for Orbital Transfer Vehicles." General Dynamics Convair Division.

3. Foust, R.R. *RL 10 Derivative Engines for the OTV.* AIAA Paper 85-1338, July 1985.

4. MacConochie, I.O., J.J. Rehder, and E.P. Brien. "Preliminary Design for a Space-Based Orbital Transfer Vehicle." *J. Spacecraft* Vol. 17 No. 3, November 1979.

5. Maloney, J.W. and L.R. Pena. *Maintaining and Servicing a Space-Based Orbital Transfer Vehicle (OTV) at the Space Station.* AIAA Paper 86-2332, September 1986.

6. Redd, L. *Main Propulsion System Recommendations for an Advanced Orbital Transfer Vehicle.* AIAA Paper 85-1336, July 1985.

Chapter 4

1. Anderson, J.D. *Introduction to Flight.* New York: McGraw-Hill, 1985.

2. Braun, W.V. and F.I. Ordway III. *Space Travel a History, an Update of History of Rocketry and Space Travel.* New York: Harper and Row, 1985.

3. Holtzman, R.T. *Chemical Rockets and Flame and Explosives Technology.* New York: Marcel Dekker, 1969.

4. Kaplan, M.H. *Modern Spacecraft Dynamics and Control.* New York: John Wiley and Sons, 1979.

5. National Office for Aeronautical Studies and Research (ONERA). *Rocket Propulsion.* Amsterdam: Elsevier Publishing, 1960.

6. Sutton, G.P. *Rocket Propulsion Elements.* New York: John Wiley and Sons, 1976.

Chapter 5

1. Heppenheimer, T.A. "Paving the Way for Space Tugs." *High Technology,* September 1985.

2. MacConochie, I.O., J.J. Rehder, and E.P. Brien. "Preliminary Design for a Space-Based Orbital Transfer Vehicle." *J. Spacecraft* Vol. 17 No. 3, November 1979.

3. Maiden, J. *Development of Design Data on an Ultra-High Modulus G/E Composite for Space Application.* 30th National SAMPE Symposium, March 1985.

4. Newman, J.C. *Fracture Analysis of Surface and Through Cracks in Cylindrical Pressure Vessels.* NASA TN D-8325, December 1976.

5. Rao, K.T. "On the Fracture Toughness of Aluminum-Lithium Alloy 2090-T8E41 at Ambient and Cryogenic Temperatures." *Scripta Metallurgica* Vol. 22 No. 1.

6. Rehder, John J. "Multiple Pass Trajectories for an Aeroassisted Orbital Transfer Vehicle." *Thermal Design of Aeroassisted Orbital Transfer Vehicles.* H.F. Nelson ed. New York: AIAA, 1984.

7. Torre, C.N. *Low-Pressure/Lightweight Cryogenic Propellant Tank Design for the Space-Based Orbital Transfer Vehicle.* AIAA Paper 86-0915, 1986.

Chapter 6

1. Haviland, J.K., D.R. Moore, R.M. O'Donnell, T.B. Clark, D.M. Martin, D.J. Trillian, J.E. Medford, M.L. Mimier, W.F. Walker, B.P. Rolfe, and J.H. Bridges. *Thermomechanics Methodology Report.* Thermomechanics Group (3-52010), 1962.

2. Incropera, F.P. and D.P. Dewitt. *Fundamentals of Heat and Mass Transfer*. New York: John Wiley and Sons, 1985.

3. Menees, G.P., K.G. Brown, J.F. Wilson, and C.B. Davies. *Aerothermodynamic Heating and Performance Analysis of a High-Lift Aeromaneuvering AOTV Concept.* AIAA Paper 85-1060, June 1985.

4. Scott, C.D. and S.M. Derry. *Catalytic Recombination/Space Shuttle Heating.* AIAA/ASME Paper 82-0841, June 1982.

Chapter 7

1. International Fuel Cells. "Orbiter Fuel Cells Fact Sheet." South Windsor CT.

2. Joels, Kerry M. and Gregory P. Kennedy. *The Space Shuttle Operator's Manual.* New York: Ballantine, 1982.

3. Manzo, Michelle A., and Mark A. Hoberecht. "A 37.5-kW Point Design Comparison of the Nickel-Cadmium Battery, Bipolar Nickel-Hydrogen Battery, and Regenerative Hydrogen-Oxygen Fuel Cell Energy Storage Subsystems for Low Earth Orbit." 19th Intersociety Energy Conversion Engineering Conference, San Francisco, 19-24 August 1984.

4. National Aeronautics and Space Administration. *Electrical Power System Workbook.* Houston: NASA, 1982.

Chapter 8

1. Faget, M.A. "Space Shuttle Life Support Systems: A Status Report." XXXII Congress International Astronomical Federation, Rome, 6-12 September 1981.

2. Haffner, J.W. *Radiation Shielding in Space.* New York: Academic Press, 1967.

3. Meness, G.P. "Design and Performance of a Conical Aerobrake Orbital Transfer Vehicle." *Thermal Design of Aeroassisted Orbital Transfer Vehicles.* H.F. Nelson ed. New York: AIAA, 1984.

4. Normansell, M. "Proposal to Allied-Bendix Corp. on SSERV: Space Station and Emergency Rescue Vehicle." Rescue and Emergency Aerospace Capabilities Team (REACT). University of Virginia, 29 April 1987.

Chapter 10

1. Byington, L., and D. Theis. "Data-Processing Hardware for Spacecraft - Air Force Standard 1750A-ISA is New Trend." *Computer* November 1986: 50-59.

2. Carney, P. "Selecting On-Board Satellite Computer Systems." *Computer* April 1983: 35-41.

3. Greenberg, E. et al. "Survey of Spacecraft Memory Technologies." *Computer* March 1985: 29-39.

4. National Aeronautics and Space Administration. *Data Processing System Overview Workbook.* Houston: NASA, 1984.

5. Ross, Cindy. *State-of-the-Art Report: Data Management Systems.* Dept. of Mechanical and Aerospace Engineering, University of Virginia, 1986.

6. Theis, D. "Spacecraft Computers: State-of-the-Art Survey." *Computer* April 1983: 85-97.

Chapter 11

1. Cager, Ralph H. et al. "Orbiter Ku-Band Integrated Radar and Communications Subsystem." *IEEE Transactions on Communications* COM- 26 (1978): 1604-1619.

2. Joels, Kerry M. and Gregory P. Kennedy. *The Space Shuttle Operator's Manual.* New York: Ballantine, 1982.

3. National Aeronautics and Space Administration. *Introduction to Orbiter Communication/Instumentation Systems.* Houston: NASA, 1984.

4. *Manned GEO Sortie Study.* NASA-Johnson Space Flight Center, 1983.

Chapter 12

1. Ballendock, Walter W., et al. "MMU Technology." *Aerospace America* Vol. 23, May 1985: 56-62.

2. Cowen, Robert C. "Doctoring Satellites: A Success Story." *Technology Review* Vol. 87, July 1984: 4-5.

3. Eberhart, Jonathan. "Shuttle 7: We Pick Up and Deliver." *Science News* Vol. 124, 2 July 1983: 4-5.

4. Minor, Ronald. *Design of a Mechanical System for Orbital Recovery of a Satellite*. University of Virginia, May 1988.

5. "New Cherry Picker Set for Orbit." *Science Digest* Vol. 91, April 1983: 22-23.

Chapter 13

1. *Thomas Register of American Manufacturers and Thomas Reister Catalog File*. Thomas Publishing, New York.

2. Lowther, William. "The Cost of the Shuttle." *MacClean's* Vol. 97, 16 April 1984: 53.

Chapter 14

1. Goldman, Nathan C. *Space Commerce: Free Enterprise on the High Frontier*. Cambridge, MA: Ballinger Publishing, 1985.

2. Paules, Granvill. "Space Station Overall Management Approach for Operations." AIAA Space Station in the 21st Century Conference, Reno, NV, 3-5 September 1986.

3. Steiner, George A. and William G. Ryan. *Industrial Project Management*. Toronto: Macmillian, 1968.

4. Stekler, Herman O. *The Structure and Performance of the Aerospace Industry*. Los Angeles: University of California Press, 1965.

Appendix 2

1. Bate, R., D. Mueller, and J. White. *Fundamentals of Astrodynamics*. New York: Dover Publications, 1971.

2. *Future Orbital Transfer Vehicle Technology Study* Vol. 2. NASA Contractor Report 3536, Boeing Aerospace Company.

Appendix 3

1. General Dynamics. "Definition of Technology Development Missions for Early Space Station Orbit Transfer Vehicle Servicing." NASA-CR-170863, 1983.

2. Martin Marietta Aerospace. "Definition of Technology Development Mission for Early Space Station Satellite Servicing." NASA-CR-171229, 1984.

3. Natqatomo, N. "Orbital Operation of Co-orbiting Spacecraft with a Space Station." AIAA TIS 3/13.

Appendix 1

System and Subsystem Weight and Power Requirement Estimates

The following pages are tables of our estimates for the weights and power requirements of various subsystems. These estimates were based on our worst case scenario. For missions other than worst case, our weights may be lower. The total weights of various subsystems are as follows:

Table A1-1

Total System Weights (Worst Case)

ECLSS	3560 lbm
Main Fuel Tanks and Supporting Structure (6 pairs)	3660 lbm
Aerobrake	2800 lbm
Command/EVA Module and Components	12680 lbm
Electronics	985 lbm
EPS	2215 lbm
RCS	3250 lbm
Engine System	1050 lbm
Crew (3)	510 lbm
MMU (2)	1280 lbm
Robot Arm	330 lbm
Flight Chairs (3)	250 lbm
Total (Dry)	32670 lbm
Propellant (Total)	125000 lbm
Total (Fueled)	157670 lbm
Payload (Max.)	24000 lbm
Grand Total	181670 lbm

Table A1-2

Mass and Power Analysis of OTV ECLSS

System	Weight (lbm)	Power (Watts)
Air Revitalization System	650	900
Thermal Control	880	900
Crew Systems (Worst Case)		
O_2 (metabolic - 2.25 lbm/man-day)	96	-
N_2	193	-
H_2O (drinking - 8.0 lbm/man-day)	336	-
H_2O (hygiene - 15.0 lbm/man-day)	630	-
Food (2.50 lbm/man-day)	105	-
Waste (1.00 lbm/man-day)	42	-
Other Components		
Freezer and Microwave	60	700
LiOH/contaminant removal cannisters	220	-
Sanitation and Hygiene	200	-
Galley	150	-
Totals	3560	2600

Table A1-3

Electrical, Avionics, and Communications Subsystem Weights and Power Consumption

Subsystem	Weight (lbm)	Power (Watts)
GNC		
GPS Receivers (2)	40	60
Stellar Tracker	40	20
IMU (2)	40	320
Ku-Band Radar	NA	NA
Total GNC	160	400
DMS		
Computers (3)	63	300
Mass Memory (2)	31	20
Displays (4)	16	80
Keyboards (3)	15	10
Data Bus Network	100	20
Instrumentation	100	50
Total DMS	325	490
Communications		
S-Band PM Radio (2)	200	700
UHF Radio	40	25
Ku-Band Radio/Radar	260	590
Total Communications	500	1315
EPS		
Fuel Cells (2)	350	NA
Ni-H battery	165	NA
EPDS (2)	100	200
Total Reactants	1600	NA
Total EPS	2215	200
RCS		
Reaction Control System	450	300
RCS fuel	2900	NA
Total RCS	3250	300
Grand Total	7710	2705

Table A1-4

Structural Component Weight Estimate

Structural Component	Weight (lbm)
Engine Quick Disconnect Plate	.100
Thrust Structure	.210
Connectors (6 sets)	.240
LO_2 Tanks (6)	.600
LH_2 Tanks (6)	1500
Tank Support Rings and Support Struts (6 sets)	.220
Command Module	10700
EVA Module	1500
Aerobrake	2800
Hatches (3)	.300
Docking/Service	.180
Total	18850

Appendix 2

Orbital Mechanics

Great emphasis was spent on determining the proper trajectory of the OTV. The biggest problem was to determine a successful rendezvous and intercept trajectory with a target satellite in geosynchronous orbit (GEO). Potential launch windows were investigated, but after careful analysis, it was discovered that only one launch window was necessary. To achieve this result, a Phase Injection Burn (PIB) was proposed [1]. Below is a thorough explanation of this maneuver.

PIB is used to phase the OTV with the GEO satellite so that an approximation to the Hohmann transfer from LEO to GEO can be executed. Consider FIGURE A2-1. What is done is that the OTV's time of flight from LEO to GEO is first determined. Then, this value is used to ascertain the angular displacement of the GEO satellite from the intercept point. Thus, in order for a successful rendezvous to occur at GEO, the target satellite must be at an angle of 79.2° from the line of nodes at the commencement of the OTV Hohmann transfer. Note that the OTV transfer can only be initiated at the line of nodes intersecting GEO and LEO. By the time that the OTV reaches the intercept point, the satellite will have traveled the 79.2° displacement.

The main problem is that the position of the OTV (point A) and the position of the satellite (point B) at the beginning of LEO-GEO transfer rarely occurs, if at all. PIB rectifies this situation.

What happens is that the OTV is launched into a round trip elliptical path from A when the satellite is found at any point on arc PC. The time of flight of the OTV's PIB corresponds to the time it wll take the satellite to reach point B. Thus, by the time the OTV returns to its original location (point A), both spacecraft are perfectly phased for intercept through the Hohmann transfer.

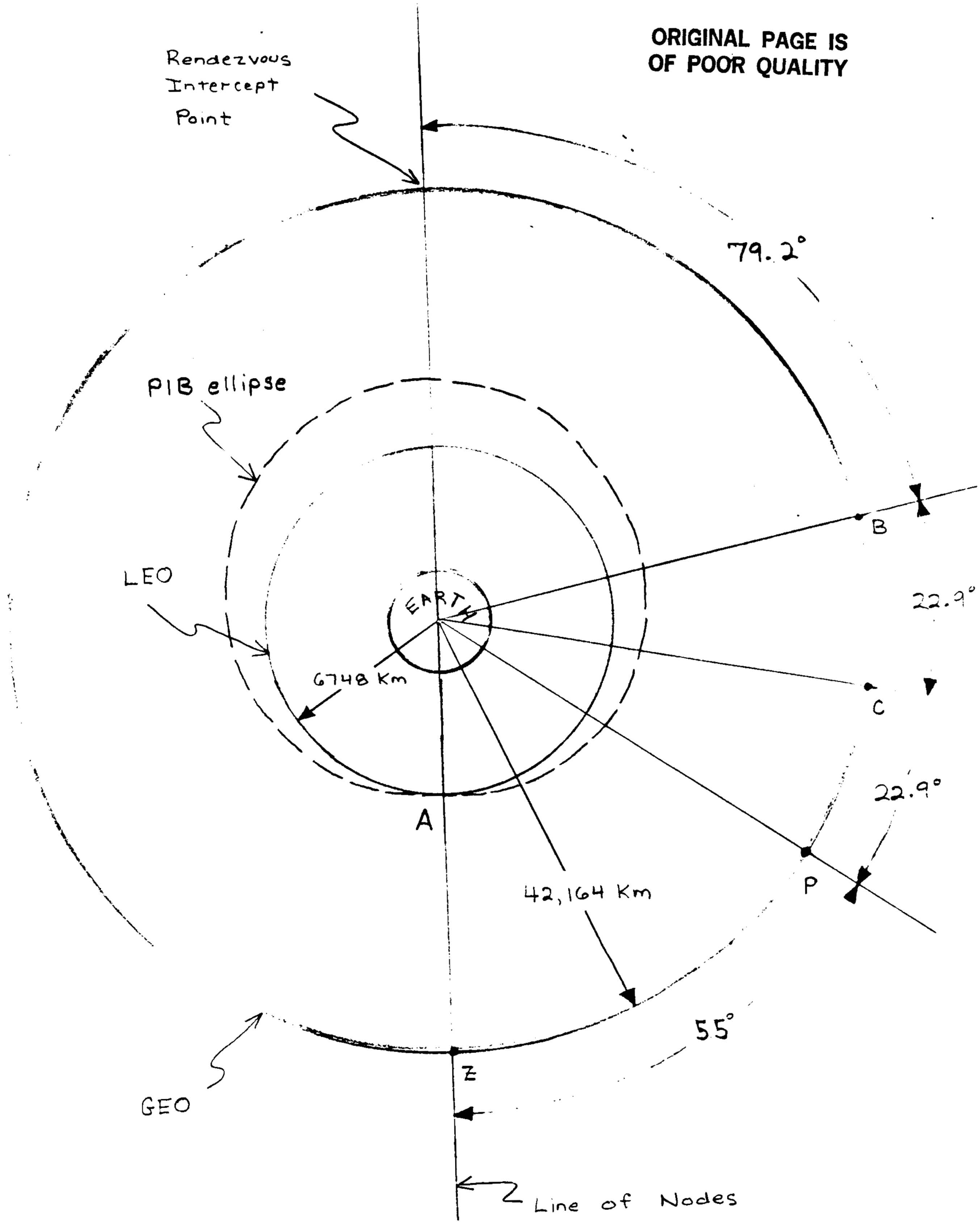


Figure A2-1: Trajectory Schematic

Under no circumstances is PIB to be executed if the satellite is on arc CB. This is because the corresponding PIB will take the OTV inside LEO where it will encounter significant drag.

Of course, it is possible to perform a PIB if the satellite is found on the arc ZC. However, such a launch would mean a waste of propellent because one complete revolution at LEO corresponds to 22.9° of GEO displacment. In other words, by the time the satellite reaches C from Z, the Space Station will have passed point A about 3 times. Using the same argument, by the time the satellite reaches P from Z, the Space Station will have passed point A about two times.

Arc PC was chosen for the delta V analysis because this displacement corresponds to one complete revolution on LEO. In essence, it is the optimum launch opportunity for PIB. PIB can also be executed for an intercept at the other end of the nodal line. Thus, 2 rendezvous intercept opportunities are guaranteed within a 24 hour period from the Space Station.

Since LEO is inclined 28.5° relative to GEO, it was found that a simultaneous plane change and circularization maneuver at GEO involved the least delta V. Furthermore, the aerobraking maneuver was thoroughly investigated. Details of this maneuver are discussed in Chapter 2.

Table A2-1 is a chart outlining the required delta V's which the OTV will need to execute for a typical mission. See Figure A2-2. For simplicity, the aero-assist trajectory is not included.

Table A2-1

Summary of Delta V's

ΔV(km/s)	LOCATION	REASON
1.5166	1	PIB
0.8893	2	Injection transfer from LEO to Hohmann transfer ellipse
1.8258	3	Circularization and plane change at GEO
1.8437	4	Plane change for LEO return and to shorten perigee height for aerobraking
0.0000	5	Aerobraking at 80 km altitude maximum. 2 passes through Earth atmosphere. (Free velocity decrement of 2.250 km/s)
0.4513	6	LEO circularization

TOTAL ΔV's = 6.5268 km/s

It is important to note that the sum of the delta V's at locations 1 and 2 (2.4059 km/s) is invariant. This means that no matter what the PIB and the transfer injection delta V's are, their sum will always equal 2.4059 km/s. Also note that the total propulsive delta V's to GEO is approximately equal to the ones needed to return to LEO (aerobraking velocity increments included).

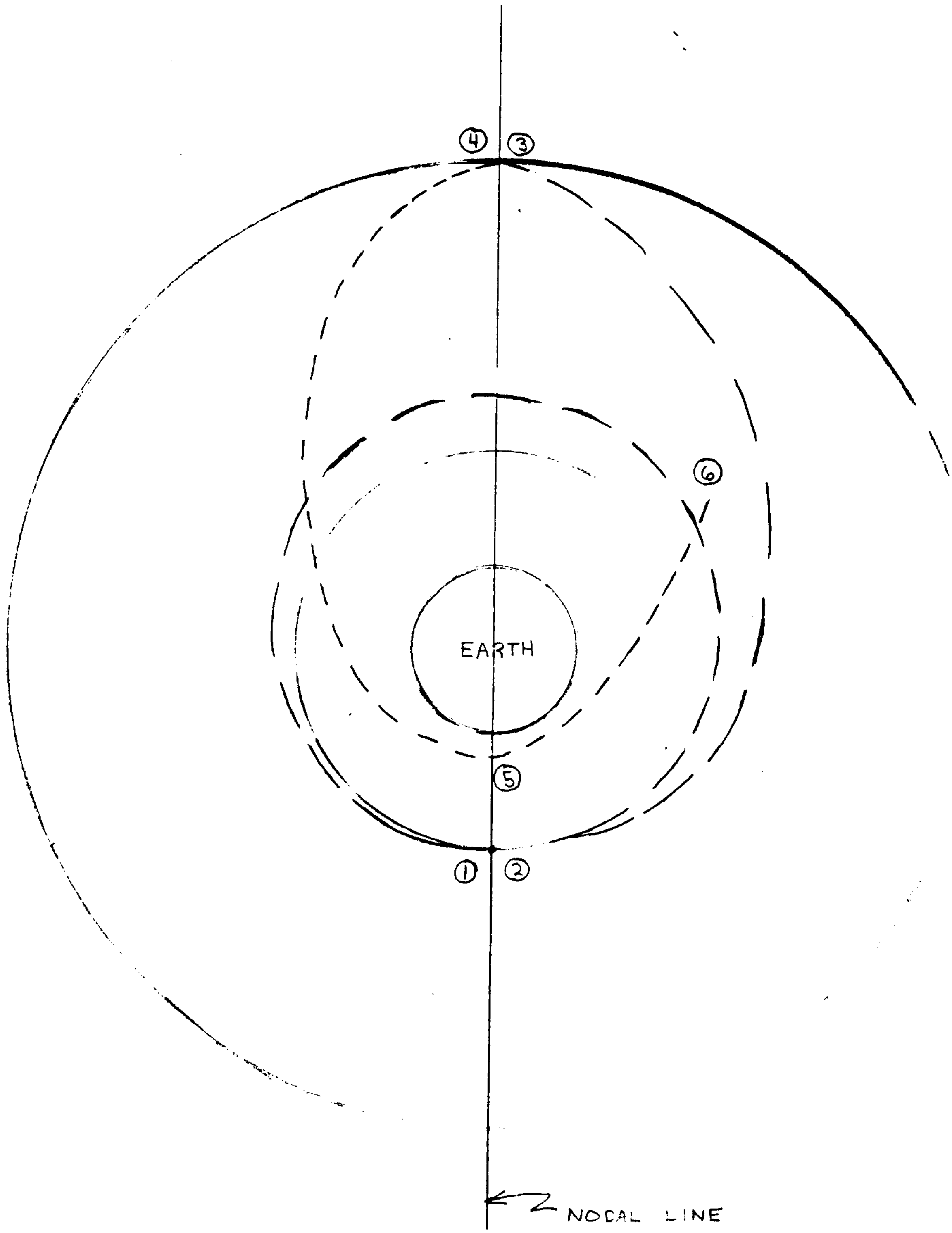


Figure A2-2: Location of OTV's Delta V's

Appendix 3

OTV Servicing Aboard the Space Station

When the Space Station becomes operational sometime in the mid-1990s, there will be a need to service OTVs. WWSR's OTV will require some servicing during the time between missions. This repair and refurbishment will take place in a special area aboard the Space Station. This area will need to be separated from the main portion of the station by some distance. This does not mean the repair area will be free-flying, only out on a boom away from the living quarters, and scientific areas.

The servicing area will mainly consist of a large hangar. This hangar will consist of several distinct areas. These areas will be: fuel depot, engine bay, fuel tank storage, cargo handling, avionics repair, heat shield repair, command module repair and refurbishing, and ship integration area.

The fuel depot will consist of several large cryogenic storage tanks for the LH2 and LO2. These fuels will be stored in insulated thermos-like tanks that will have to be small enough to carry up in the shuttle cargo bay. These tanks will have to be protected from the rays of the sun, as well as have protection from being punctured by meteorites. The protection from the Sun will consist of moveable shades that will move as the direction of the Sun changes. The protection of the tanks from puncture will consist of a honeycomb structure that will stop all but the largest meteorites. If a tank happens to get punctured by a large object, it will have to be jetisoned immediately so that the escaping gases do not create a force imbalance on the Space Station.

The engine bay of the servicing area will be the place where spare engines for the OTV are stored and repaired. The OTV engines will be modularized, so they will only need to be snapped on and off between missions. The engines may be taken out after each sortie to make sure that no malfunctions happen during a mission. In the event that the engine cannot be repaired in space, it will have to

be brought down to Earth on the Space Shuttle. Part of the engine bay will consist of the storage empty spare fuel modules for the OTV. For the moving of these tanks, as well as the engines and other large portions of the OTV, the hangar will have a large servicing crane. This crane will be on a track that will run down the length of the hangar, and will have enough power to move the whole assembled OTV.

The cargo handling area of the hangar will consist of a place to store the satellites before they are loaded on the OTV for transfer to GEO. The satellites may have to wait long periods of time before they can get a flight out to GEO. This means that the satellites are able to be checked out and serviced in this waiting area. The cargo area will also need a means of transferring the retrieved satellites from the OTV area to the satellite repair area.

The repair area of the hangar is probably the most important. The station must be able to repair all but the most severe malfunctions without having to send portions of the OTV back to Earth. This will mean that there must be astronauts on the station that are knowledgeable in all areas of the OTV, and that the repair area will be equipped well enough for repair of all major portions of the OTV including: avionics, life support, reaction control engines, fuel handling, cargo handling, and heat shield.

Finally, the ship integration area is where the whole OTV will be assembled. This area will need to be large enough to contain an entire assembled OTV. The integration area will need to have several cranes, as well as robot arms for the astronauts to stand on while putting the OTV together. This is also where the OTV will be stored between missions. The reason for storing the OTV inside the hangar is to protect it from damaging radiation, micrometeorites, and random debris that will be floating around the Space Station.

The cost of this repair satation has yet to be determined because the area has yet to be fully designed. The current estimates are that the area will cost about $700 million. This does not include the cost of sending up parts on the

shuttle. This repair hangar and all attached areas will take about three shuttle flights to lift to LEO. At current costs, this means another $300 million to the price for a grand total of $1.0 billion. This price is only preliminary and will no doubt increase as production of the pieces moves ahead.

Appendix 4

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